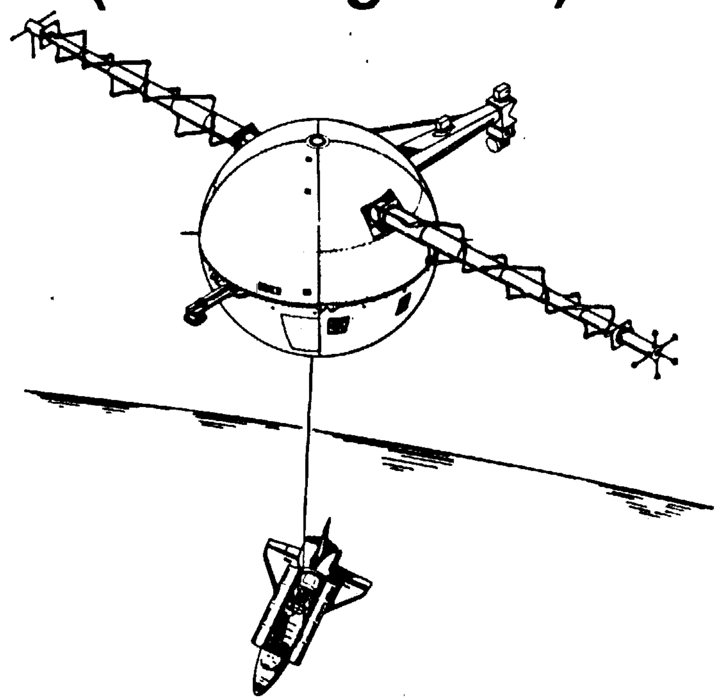
Tethered Satellite System (TSS-1)
(STS Flight 46)



Volume II-System Description

MARTIN MARIETTA

Purpose of Document - Volume II

This Document Has Been Prepared to:

1) Provide an In-Depth System Description for Technical Reviews and Briefings By NASA and Martin Marietta Astronautics Group (MMAG) Personnel.

2) Provide TSS-1 Familiarization Training for New Personnel Who Are Assigned to the NASA and MMAG Project for Future Missions.

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1. System Requirements

System Requirements

Description

The TSS is a Reusable Orbital Flight Facility for a Variety of Scientific Experiments and Investigations in Near Earth Orbit.

The TSS Consists of:

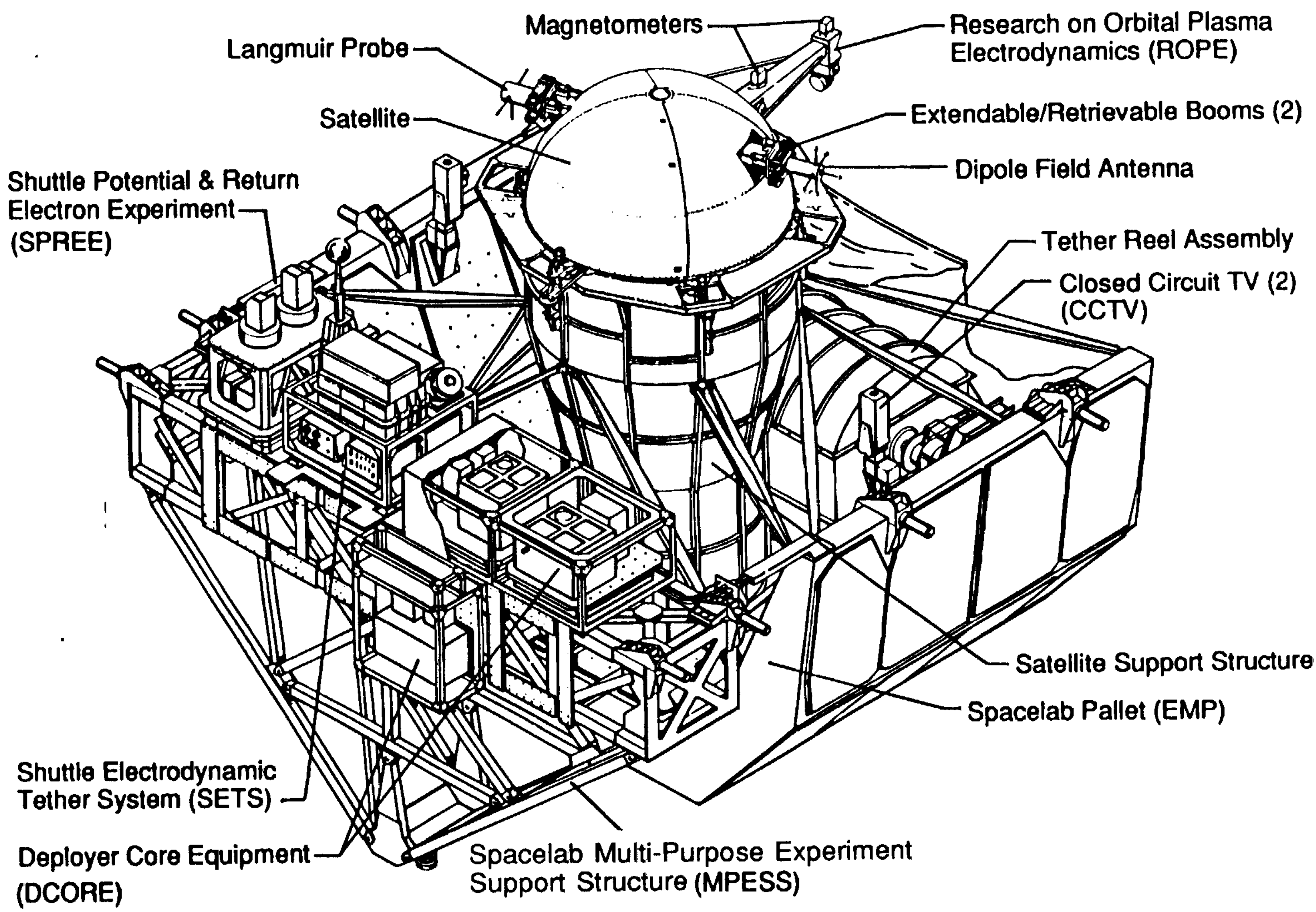
Deployer -	Orbiter Mounted Hardware and Software Required to Secure the Satellite During Launch and Landing, and Deploy and Retrieve the Satellite
Satellite -	An Instrument Support Module, Containing the Mission Instruments and a Service Module, Containing the Satellite's Supporting Subsystems
Instruments -	Various Sensory Devices, Selected for each Mission, Which Perform Specific Mission Scientific Objectives

The TSS Specifications are:

System	TSS-CEI-01 -	Project Specification, Performance, Design and Verification Requirements for Tethered Satellite System
Deployer	TSS-CEI-02 -	Deployer CEI specification, Performance, Design and Verification Requirements for the Tethered Satellite System
Satellite	TS-SY-AI-001 -	TSS-S Specification

System Requirements

TSS Hardware



System Requirements

Missions

The TSS is Capable of Deploying, Maintaining On-Station, & Retrieving a Satellite from the Orbiter Cargo Bay Either Towards or Away from the Earth.

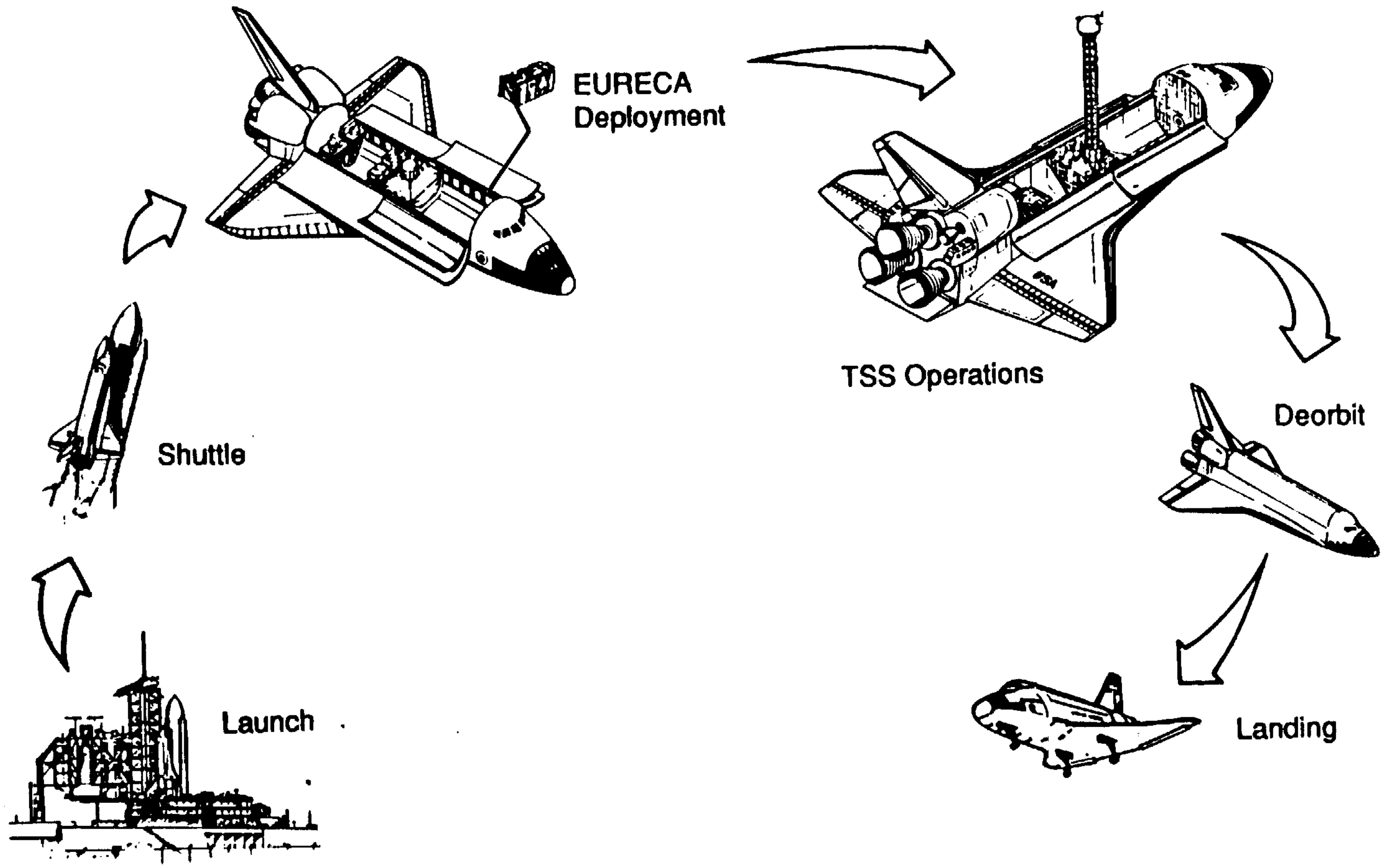
For Electrodynamic Missions, the TSS is Capable of Deploying a Satellite, on a Conducting Tether, to a Distance of 20 km Away from the Earth.

For Atmospheric Missions, the TSS is Capable of Deploying a Satellite to Distance of 100 km Towards the Earth.

The TSS is Designed to be Compatible with the Nominal STS Mission Duration of 10 Days.

System Requirements

Missions



System Requirements

Interface Requirements

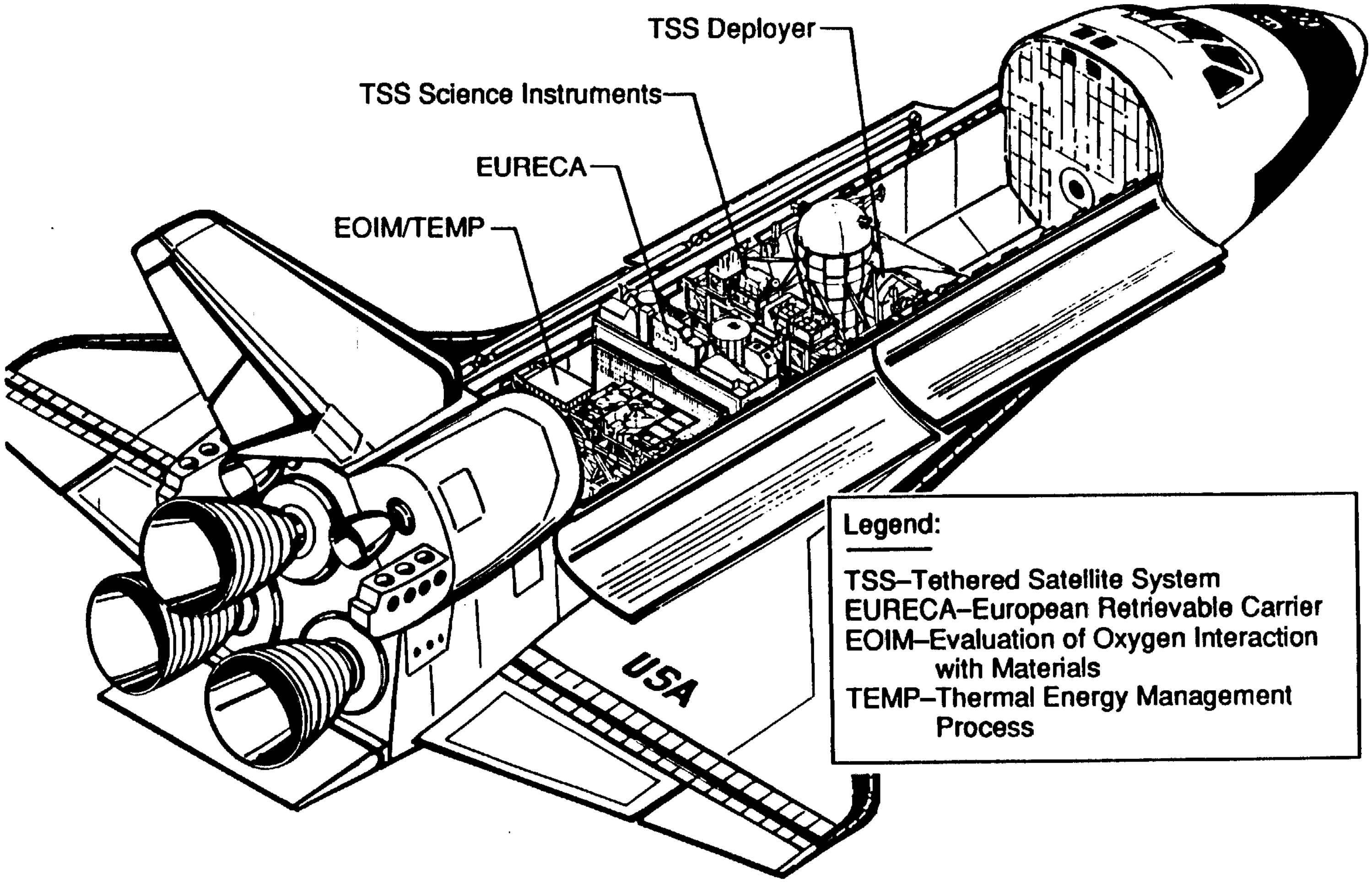
The TSS is Compatible with the Orbiter Cargo Bay Interface Requirements as Defined in ICD 2-19001.

The TSS Utilizes a Spacelab Enhanced MDM Pallet as Modified for TSS Usage and is Compatible with the Mission Specific Interface Requirements Defined in ICD-A-18411 and the MDM Pallet Interface Requirements as Defined in ICD-B-18411.

The TSS is Compatible with the Interface Requirements Defined in SPAH SLP 2104.

System Requirements

Payload Bay Integration



Legend:

TSS–Tethered Satellite System
EURECA–European Retrievable Carrier
EOIM–Evaluation of Oxygen Interaction with Materials
TEMP–Thermal Energy Management Process

System Requirements

General Performance

TSS Launch Operations Shall be Compatible with STS Operations at Pre-Launch, Launch, Landing and Post-Landing.

The System Shall Provide Status Monitoring during Pre-Operational and Post-Operational Mission Phases.

The System Shall Provide Status Monitoring and Crew Control During All Operational Mission Phases.

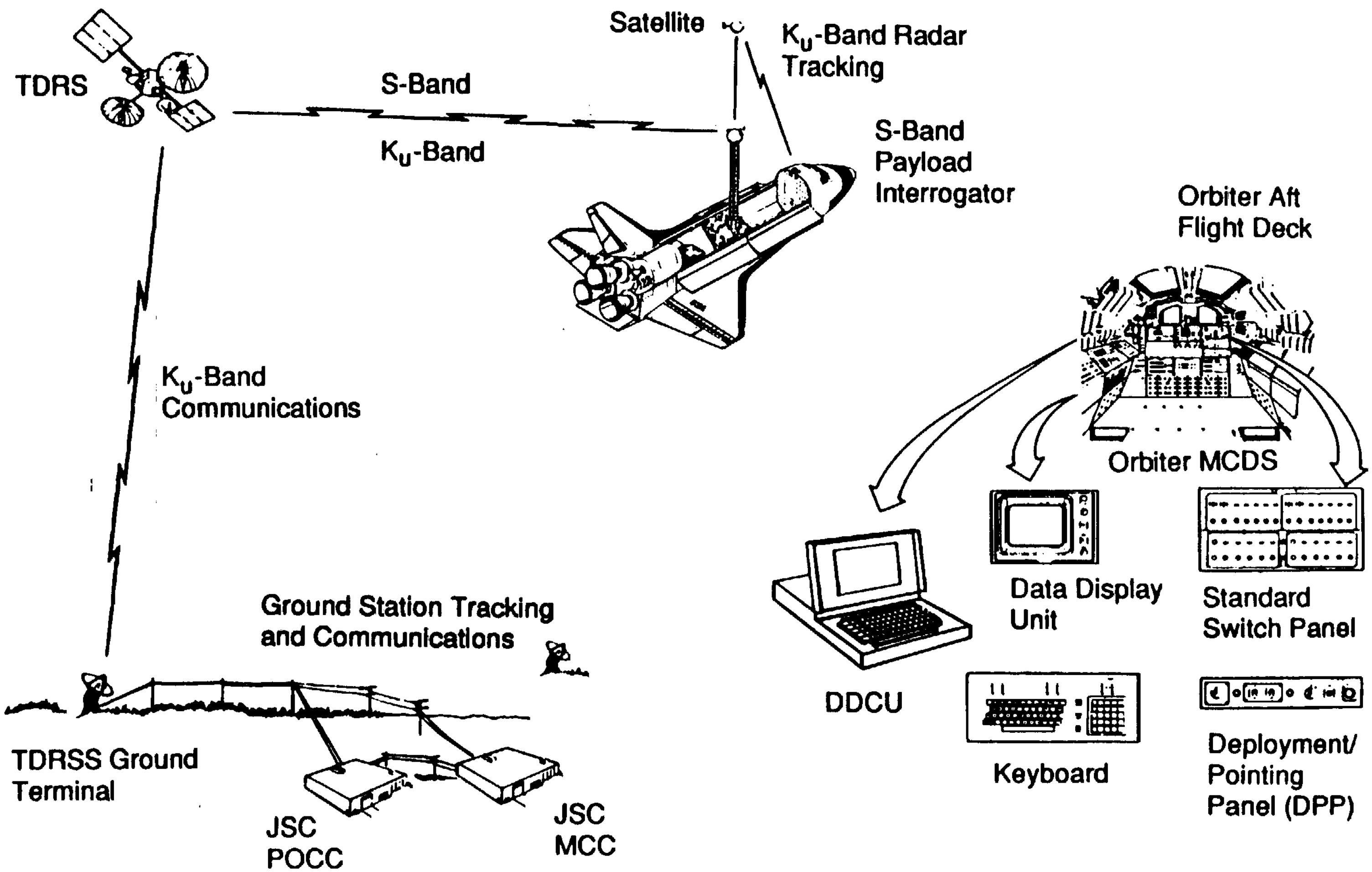
The System Shall maintain Satellite Control and Stability During Deployed Operations by Utilizing Deployer Tether Feedback and Satellite Thrusters.

Satellite In-Plane and Out-of-Plane Angular Deflections are Controlled by Crew Commands to the Satellite Thrusters and Orbiter Maneuvers.

The System Shall Allow for Stopping and Restarting Satellite Deployment or Retrieval.

System Requirements

Operations and Control



System Requirements

General Performance

The Satellite Shall Be Positioned for Deployment with an Extendable/Retractable Boom.

The System Shall Be Capable of Rotating the Satellite about its Z-axis while Docked.

Satellite Deployment Velocity Shall be Controlled by the Deployer until On-Station Altitude is Achieved.

The System Shall be Capable of Operating with Tether Rates between ±4.5 m/s.

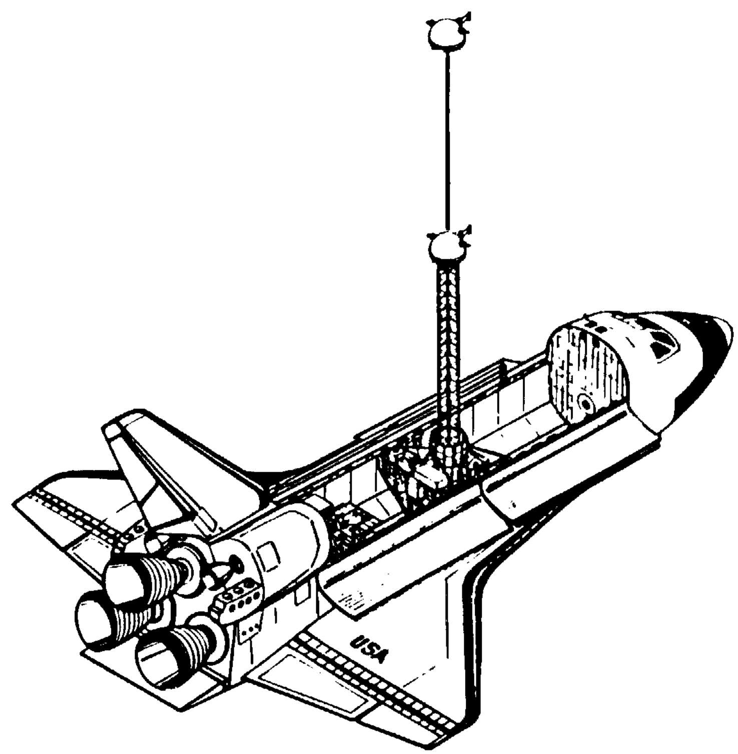
During On-Station Operations, Satellite Control Shall Be Maintained by a Combination of Passive and Active Means.

Satellite Position and Rates Are Determined Throughout the Mission By a Combination of Orbiter Radar and Deployed Tether Length.

After Docking, the Boom is Retracted and the Satellite is Restrained with the Hold-down Mechanism.

System Requirements

Operations



2. TSS Equipment Carriers

TSS-to-Orbiter Equipment Carriers

TSS-to-Orbiter Interfaces

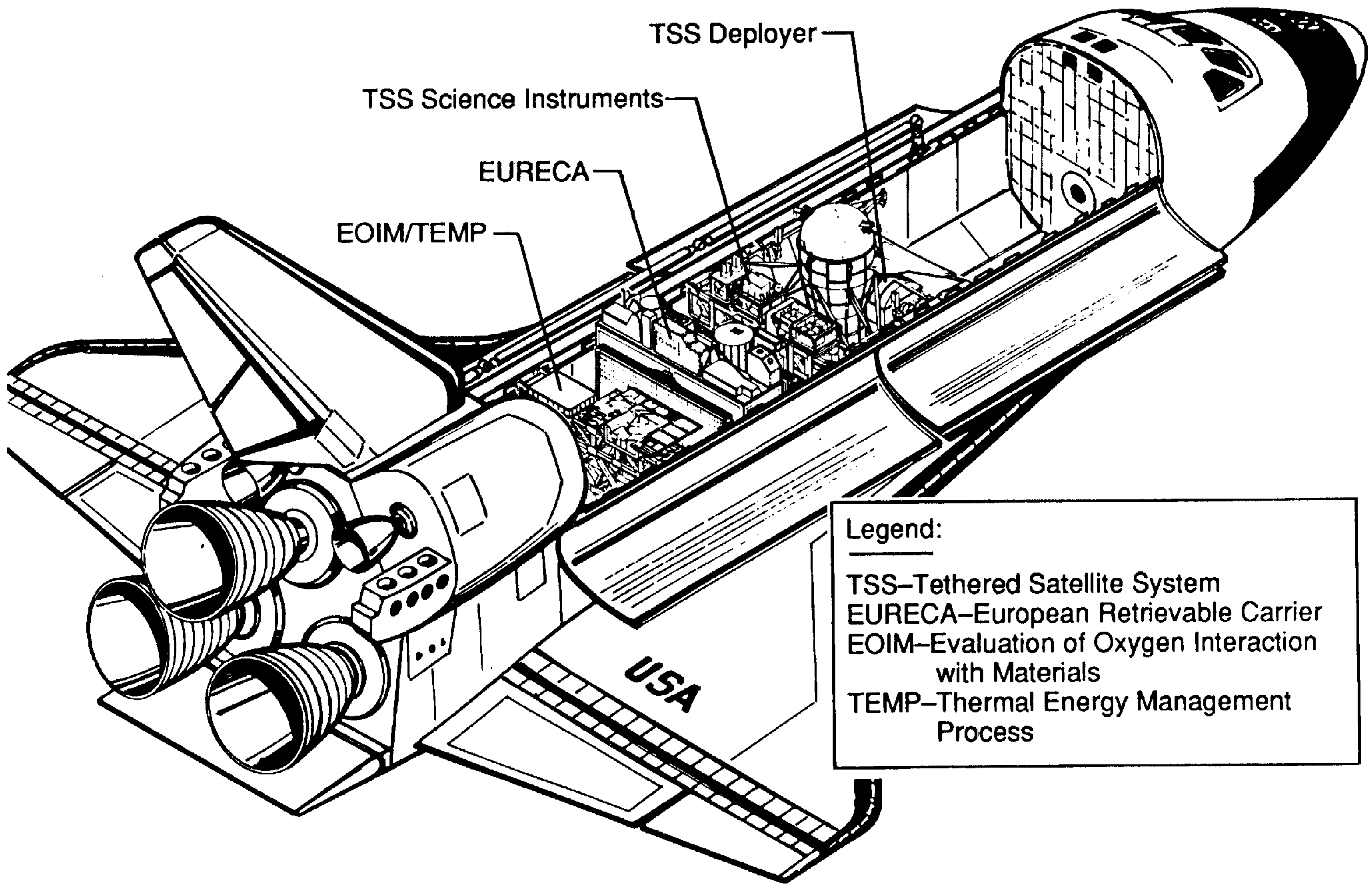
The TSS-1 Satellite, Scientific Instruments, and Supporting Hardware for the STS-46 Mission are Integrated into the STS Orbiter Cargo Bay Using (2) Equipment Support Carriers. This Mission is Shared with 2 Other Non-Related Payload Carriers. The Total Available STS On-Orbit Mission Time and Orbiter Resources Such as Power, Thermal Control and Data Management are Shared Between All of the STS-46 Payloads.

Requirements for the TSS-To-Orbiter Interfaces are Controlled by NASA/JSC Documents:

NSTS-18411	Tethered Satellite System Payload Integration Plan, and Its Related Annexes and ICS's
JSC-07700 Vol. XIV	Space Shuttle Payload Accommodation
ICD-2-19001	Shuttle Orbiter/Cargo Standard Interfaces

TSS-to-Orbiter Equipment Carriers

TSS-to-Orbiter Interfaces



TSS-to-Orbiter Equipment Carriers

Tethered Satellite System

The Tethered Satellite System for STS-46 Mission is Mounted on 2 Major Equipment Carriers: (1) The Enhanced Multiplexer-Demultiplexer Pallet (EMP); and, (2) A Spacelab Multi-Purpose Experiment Support Structure (MPESS). The Equipment is Divided Such that the Basic Satellite, and All Tether Control Hardware is Mounted on the EMP, and All TSS-1 Mission Dedicated Scientific Instruments Are Integrated on the MPESS. These Science Instruments Operate in Conjunction with Complimentary Instruments Located in the Satellite.

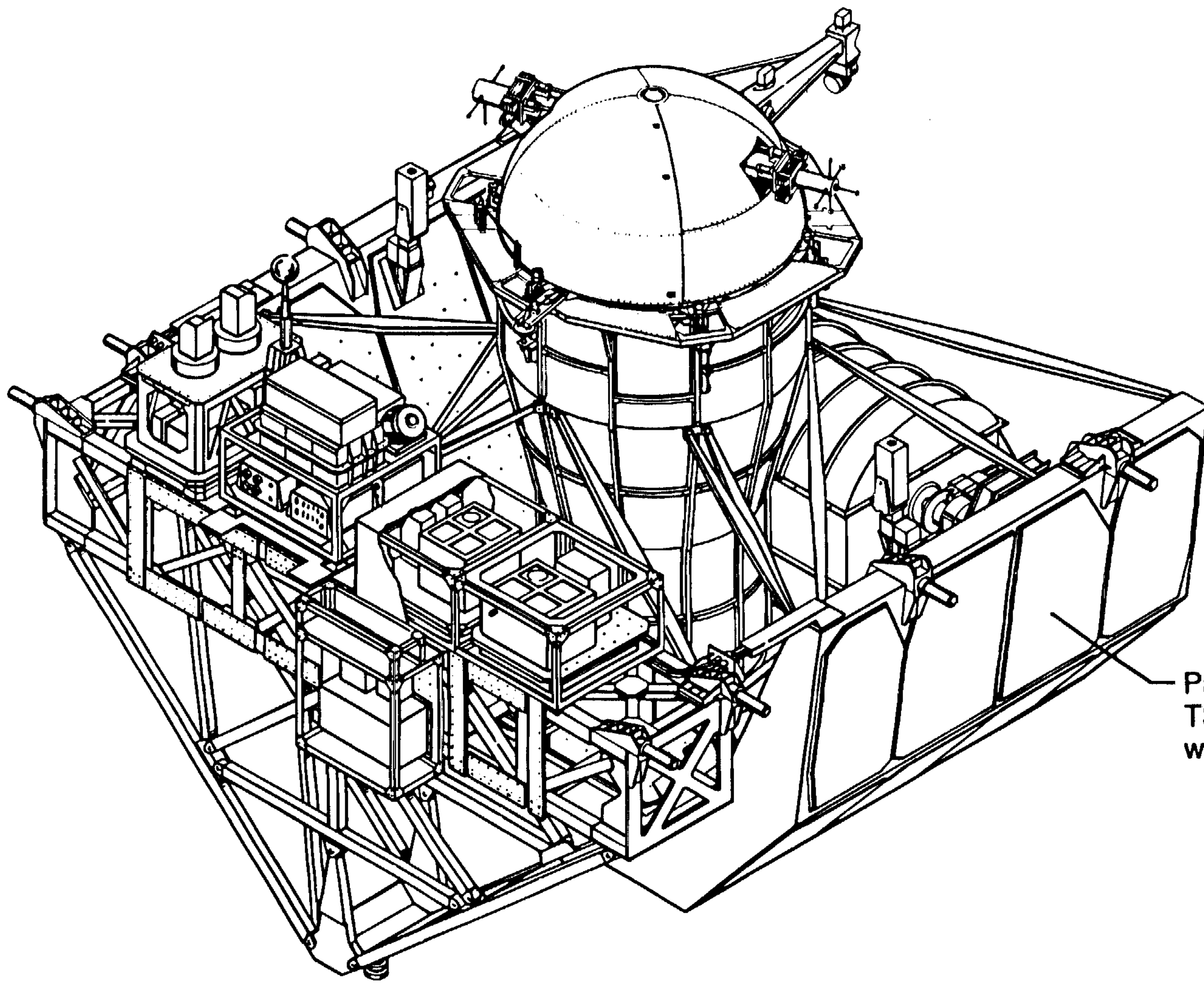
Although the 2 Equipment Carriers are Structurally Independent in their Respective Mounting in the Orbiter Cargo Bay, Interconnecting Cabling and Freon Tubing Facilitate Sharing of the Power, Data, and Environmental Control Resources for the Various TSS Subsystems and Components.

The Primary Top Level Documents Which Control the Equipment Interfaces to the Carriers and Orbiter Include:

SLP/2104	Spacelab Payload Accommodation Handbook (SPAH) and its Related Appendices
ICD-A-18411	Shuttle Orbiter/Tethered Satellite System (TSS-1) Interfaces
ICD-B-18411	Enhanced MDM Pallet/TSS-1 Interfaces

TSS-to-Orbiter Equipment Carriers

Tethered Satellite System



TSS-to-Orbiter Equipment Carriers

Enhanced Multiplexer-Demultiplexer Pallet

The Enhanced Multiplexer-Demultiplexer Pallet (EMP) is a Modified Version of the European Space Agency (ESA) Spacelab Pallet. It Consists of the U-Shaped Aluminum Structure Configured with Removable Panels and the Enhanced Thermal, Electrical Power Distribution, and Command and Data Management Subsystems. An Optional High Data Rate Multiplexing and Recording Subsystem Can Be Integrated on the Pallet if Required by the Payload. This Capability is not Required for the TSS.

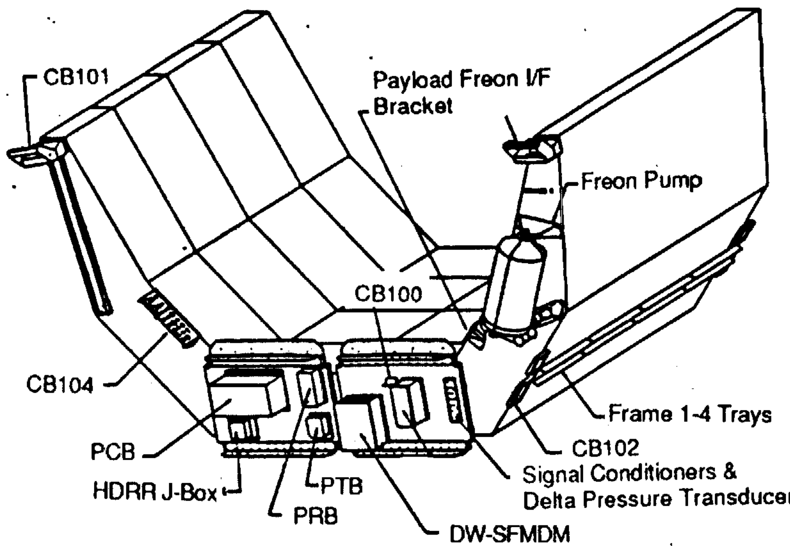
The EMP is a General Purpose Carrier Designed to Accommodate a Wide Variety of Payloads. The EMP Can Be Configured to Accommodate the Specific Needs of a Payload Such as the TSS. The Load Carrying Capability of the EMP is ≈ 3000 Kg.

Top level document which describes the EMP is:

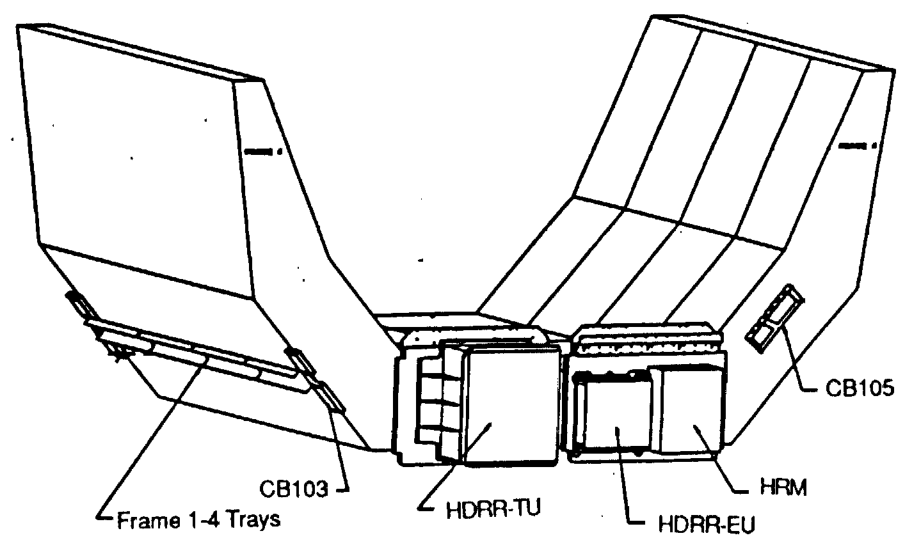
SLP/2104-7	Spacelab Payload Accommodation Handbook, Appendix G, Enhanced MDM Pallet Carrier

TSS-to-Orbiter Equipment Carriers

Enhanced Multiplexer-Demultiplexer Pallet



DW-SFMDM	Dual Wide-Smart Flexible Multiplexer/Demultiplexer.
HDRR J-Box	High Data Rate Recorder J-Box
HDRR-EU	High Data Rate Recorder Electronics Unit
HDRR-TU	High Data Rate Recorder Tape Unit
HRM	High Rate Multiplexer
PCB	Power Control Box
PTB	Payload Timing Buffer
PRB	Power Relay Box



TSS-to-Orbiter Equipment Carriers

EMP Active Thermal Control Subsystem

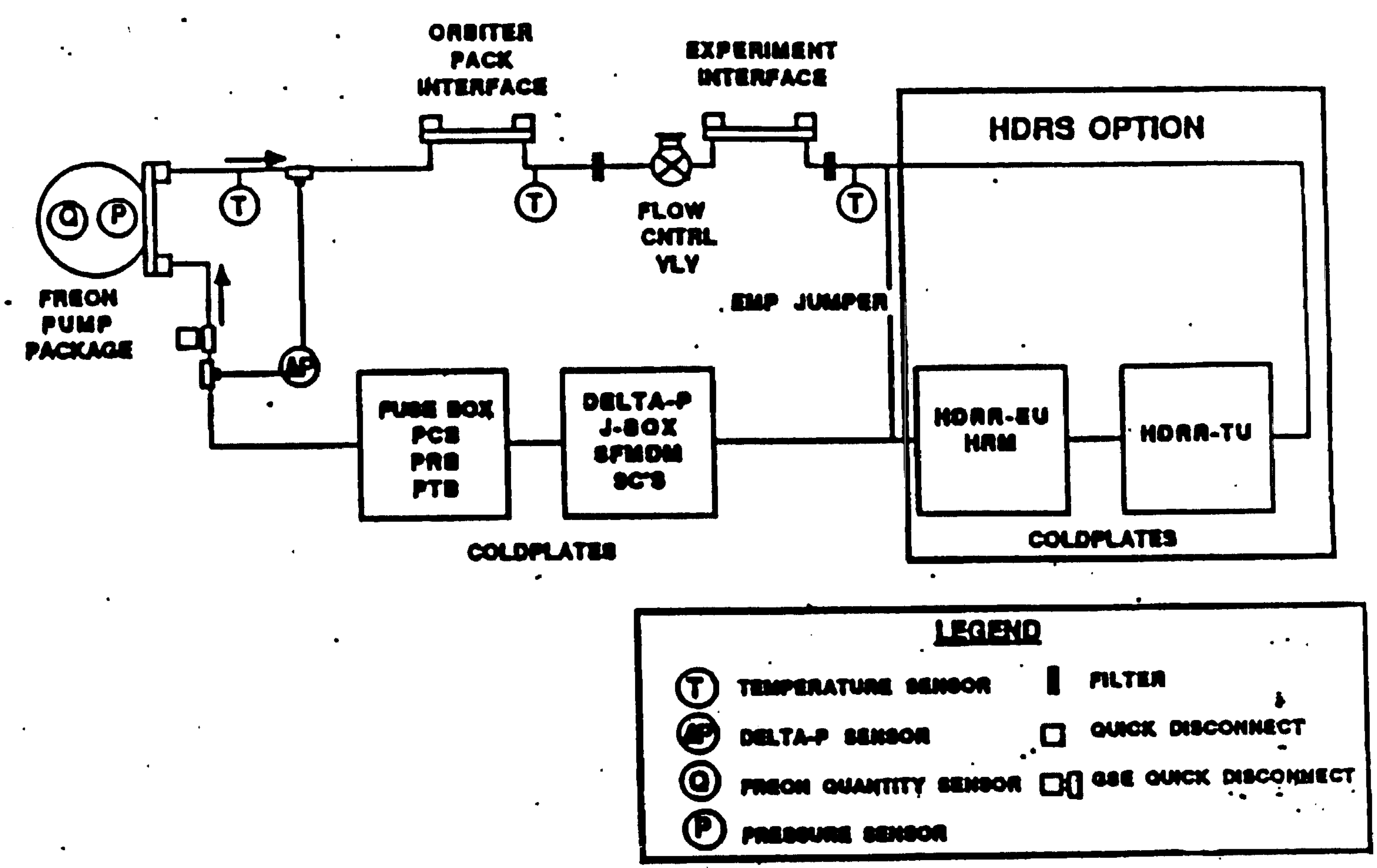
The EMP Active Thermal Control Subsystem (ATCS) Consists of a Single Liquid Phase Freon 114 Coolant Loop that Circulates Between the EMP Subsystems, Payloads, and the Orbiter Cargo Bay Heat Exchanger. The Freon Loop Can Operate Up to a Maximum Pressure of 200 psia and Flow Fates Up to 1450 Kg/hr. (3200 lbs/hr.) as Controlled by the Pallet-Mounted Pump. 3 Types of Cold Plates (ESA Standard, EMP Standard, Development Flight Instrumentation) can Be Provided for the Payloads Depending on Size and Capability Requirements. Approximately 11 Payload Cold Plates can Be Accommodated on the EMP in Addition to the EMP Support Equipment Cold Plates.

Top Level Documents Which Describe the EMP ATCS and Interfaces Are:

SLP/2104-7	Spacelab Payload Accommodation Handbook, Appendix G, Enhanced MDM Pallet Carrier
ICD-B-21084	EMP/To Payload Generic Interfaces

TSS-to-Orbiter Equipment Carriers

EMP Active Thermal Control Subsystem



TSS-to-Orbiter Equipment Carriers

EMP Electrical Power and Distribution Subsystem

The EMP Electrical Power and Distribution System (EPDS) Provides Orbiter Power and Signal Distribution, Control, and Power Circuit Overload Protection to the EMP and Payload Equipment. The Payload Power Interfaces are at Connector Brackets (CB's) CB103 and CB104. Available Circuits Include:

CB103 -	Five, 7.5 amp, Main D.C.
CB104 -	Twelve, 7.5 amp and two, 93-amp Main DC
	Two, 7.5 amp, Auxiliary DC
	One, 3-amp, 3 phase, 400 cps

Overload Protection is Provided by 20 Amp Fuses and a 100 Amp Fuse.

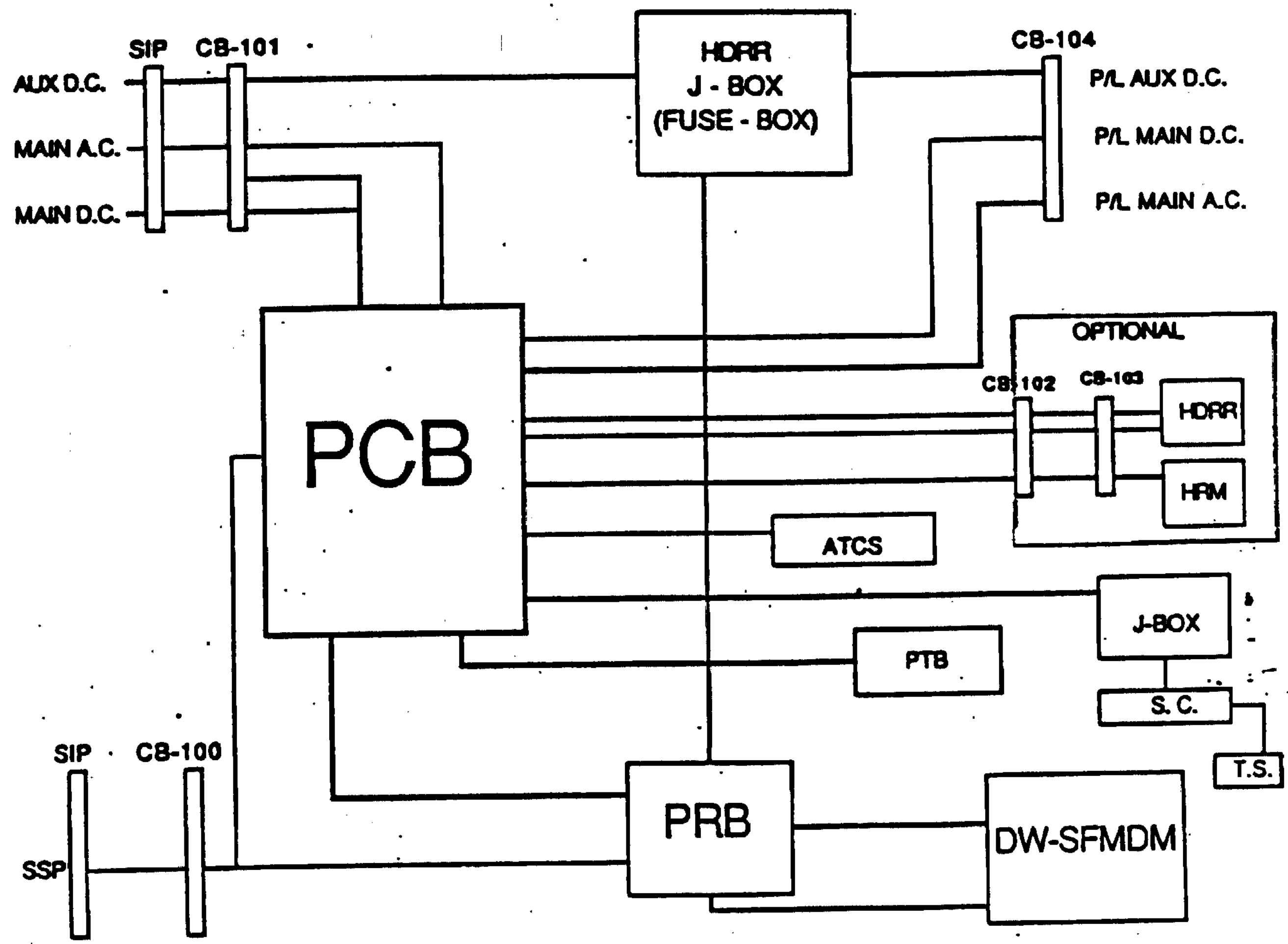
The Orbiter Standard Switch Panel (SSP) is Used to Initialize Primary Power to the EMP EPDS. Power Switching to Each of the Payload Circuits is Controlled by Commands from the EMP Avionics. Total Power Available to the EMP is 1.75 kw Maximum Continuous and 3.0 kw Peak While on Orbit. An Optional 2nd Primary Power Feeder from the Orbiter can bBe Implemented Which Will Provide an Additional 3.0 kw.

Top Level Documents Which Describe the EMP EPDS and Interfaces Are:

SLP/2104-7	Spacelab Payload Accommodation Handbook, Appendix G, Enhanced MDM Pallet Carrier
ICD-B-21084	EMP/To Payload Generic Interfaces

TSS-to-Orbiter Equipment Carriers

EMP Electrical Power and Distribution Subsystem



TSS-to-Orbiter Equipment Carriers

EMP Command & Data Management Subsystem

The EMP Command and Data Management Subsystem (CDMS) Provides Commands and Control Information to the EMP Subsystem Components and to the Payloads. It Receives and Conditions Output Status and Data from these Units: Orbiter Interfaces with the CDMS Include:

— Orbiter Payload Signal Processor (PSP) Uplink
— Orbiter Payload Data Interleaver (PDI) Downlink
— Orbiter State Vector and Attitude Data (to Payloads)
— Greenwich Mean Time (GMT) and Mission Elapsed Time (MET) Timing
— Crew Interface to Subsystems and Payloads Via the DDCU

The CDMS Dual-Wide Smart Flexible Multiplexer/Demultiplexer (SFMDM), in its Standard Configuration, can Provide:

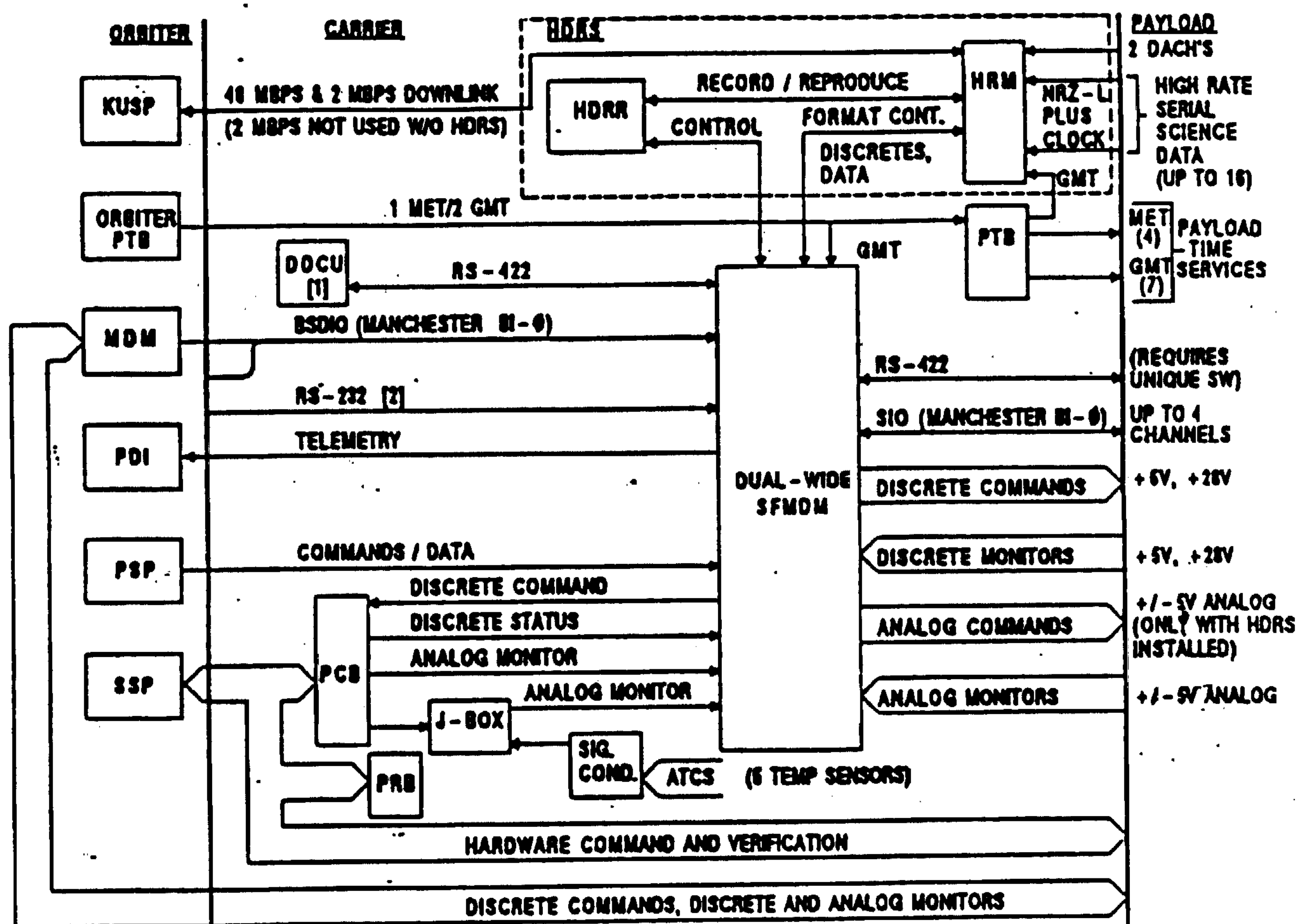
— 48 Discrete Input Lows (DIL)
— 28 Discrete Input Highs (DIH)
— 44 Discrete Output Lows (DOL)
— 26 Discrete Output Highs (DOH)
— 16 Direct Current Inputs (DCIN)
— 4 Serial Digital Input/Output (SDIO) Channels
— 1 RS-422 Channel

The DW-SFMDM is Modular in Design, and Variations in the Standard Configuration can Be Provided. The Primary Top Level Document Which Controls the Equipment Interface is:

ICD-B-18411 Enhanced MDM Pallet/TSS-1 Interfaces

TSS-to-Orbiter Equipment Carriers

EMP Command & Data Management Subsystem



[1] RESIDES ON ORBITER AFD OR MID-DECK
[2] GROUND USE ONLY

TSS-to-Orbiter Equipment Carriers

MPESS Carrier Configuration

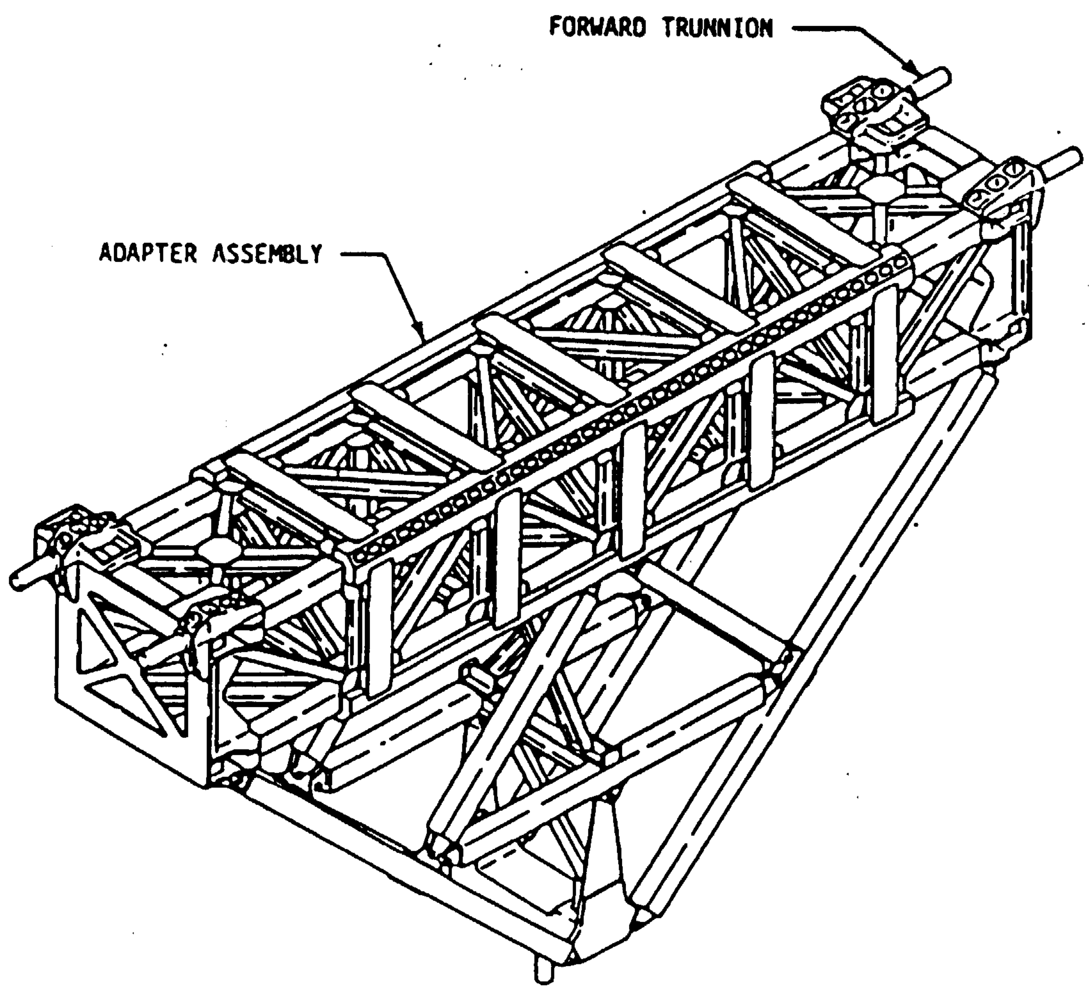
The Multi-Purpose Experiment Support Structure (MPESS) is a General Purpose Carrier Designed to Provide a Mounting for Smaller Payloads to Be Integrated into the Orbiter Cargo Bay. It Consists of a Truss Structure that Interfaces with the Orbiter Via lLongeron and Keel Trunnions. Experiment Hardware can Attach Directly to the MPESS Structure or Use an MPESS Adapter Structure Which Provides a Standard Mounting Hole Pattern for Experiment Attachment. All Thermal Electrical Power, and Data Handling Resources Must be Provided by the MPESS User through External Interfaces to the Orbiter.

Top Level Focuments Which Describe the MPESS Include:

SLP/2104-8, App. H	Spacelab Payload Accommodation Handbook Document, Appendix H, MPESS Carrier
ICD-B-MPESS	MPESS to Payload Generic Interfaces

TSS-to-Orbiter Equipment Carriers

MPESS Carrier Configuration



3. Deployer

Deployer

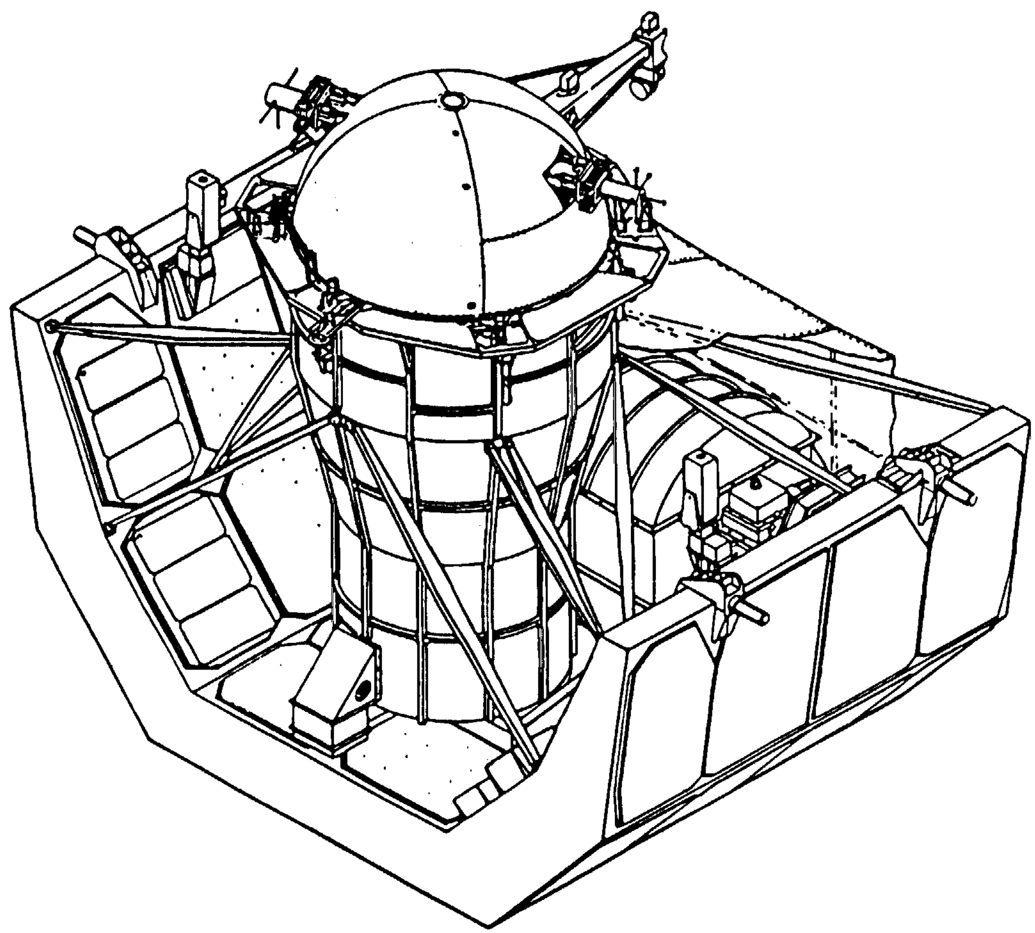
Introduction

The TSS Deployer is Designed to Perform the Basic Functions of Satellite Deployment, Maintaining the Satellite On-Station, and Satellite Retrieval.

The Deployer Consists of the Following Major Subsystems:

- Structures and Mechanisms
- Thermal Control Subsysem
- Electrical Power & Distribution
- Command & Data Management

Deployer
Introduction



Reel Assembly

Reel Support Structure & Reel

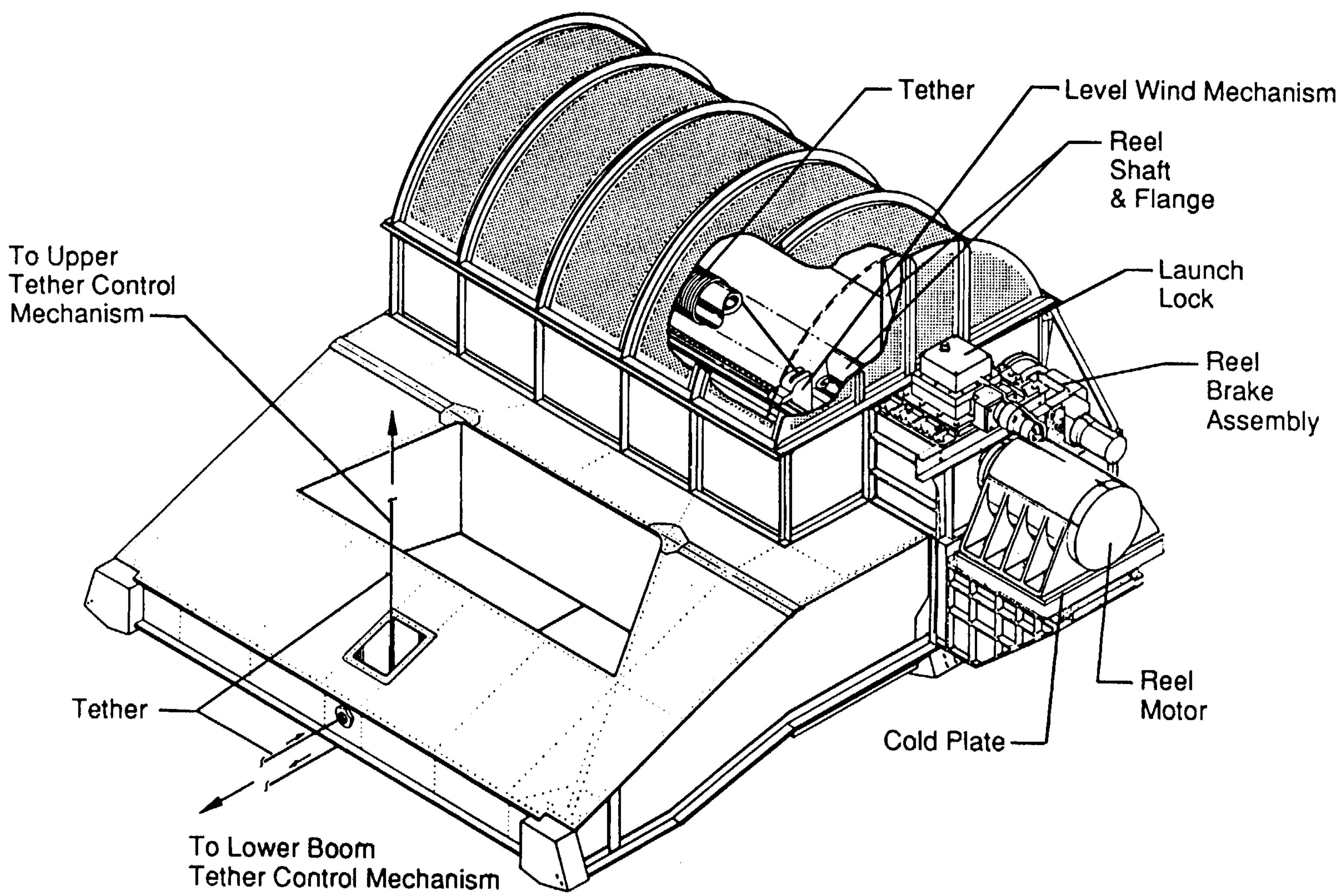
The Tether Reel Assembly Consists of the Reel Support Structure, the Reel, the Reel Motor, the Level Wind Mechanism, the Reel Launch Lock, the Reel Brake Assembly, the Slip Ring Assembly, and the Tether.

The Reel Support Structure Provides Structural Support and the Mechanical Interface between the Components Listed Above and the Pallet. The Lower Assembly Supports the Structure at Seven Pallet Hardpoints. The Upper Assembly Provides a Cover Over the Reel and Tether Containment Enclosure from the Level Wind Mechanism to the Base of the Satellite Support Assembly. This Enclosure Accommodates the Tether Sweep During Nominal Operations and Contains any Loose Tether in Off Nominal Situations. The Reel Motor Support Bracket is Attached to the Starboard Side of the Reel Support Structure.

The Reel is a 0.1128 m (4.44 Inch) Diameter by 1.2192 m (48 Inch) Wide Shaft with 0.9652 m (38 Inch) Diameter Flanges on Each End. The Reel Can Accommodate a Variety of Tether Dimensions and Lengths, Including Up to 110 Km of 2.54 mm (0.1 Inch) Diameter Tether. The Starboard end is joined to the Reel Motor and Brake by a Coupling and the Port End is Contains the Timing Gearbox to Drive the Level Wind Mechanism. The Reel is Coated to be Nonconductive for Electrodynamic Missions

Reel Assembly

Reel Support Structure & Reel



Reel Assembly

Reel Motor

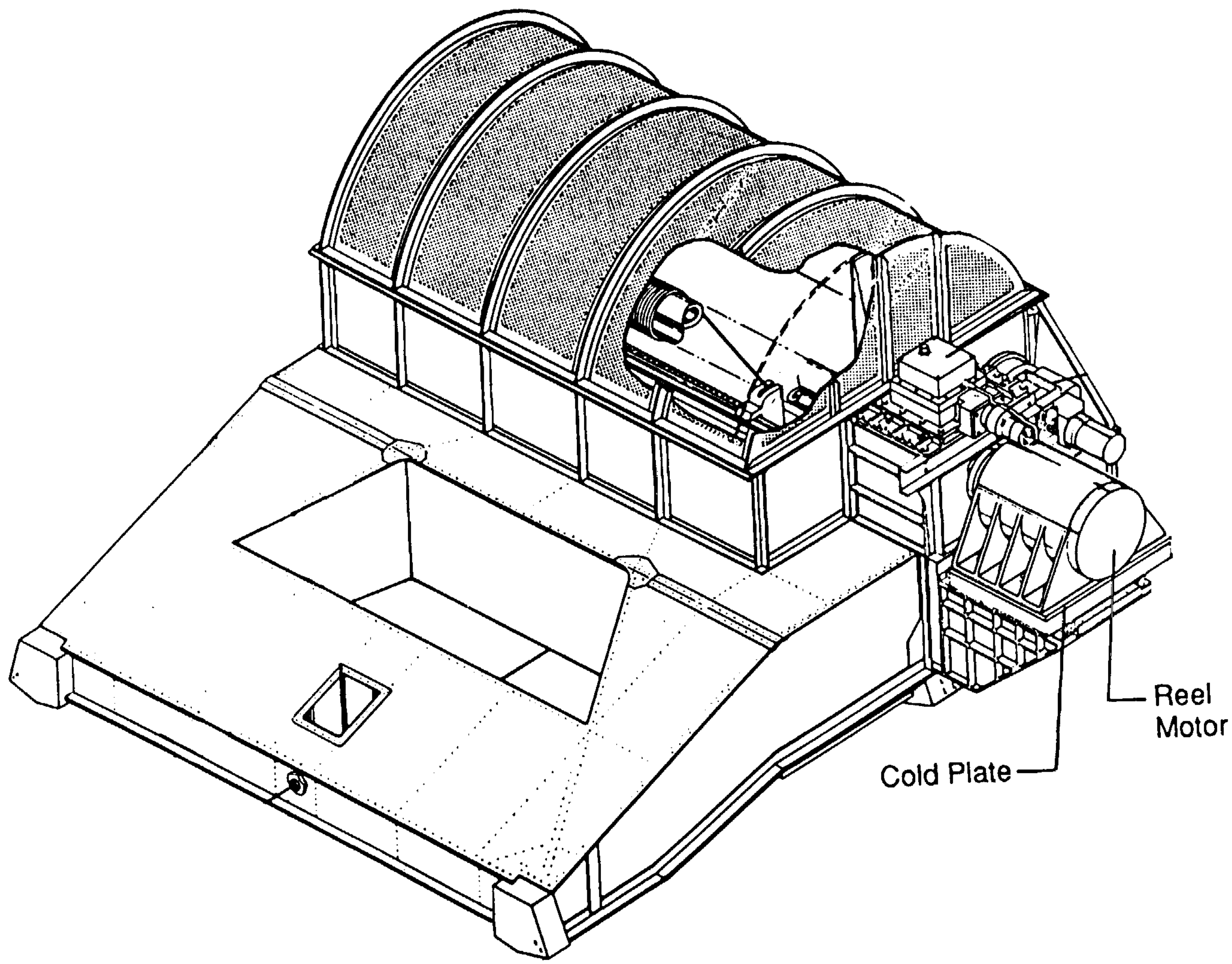
The Reel Motor is Operated as a Generator to Provide Braking Torque During Deployment, and as a Motor During Retrieval Operations.

The Reel Motor is a 3-Phase, Torque-type, Brushless Permanent Magnet Motor. It is Capable of Supplying Up to 54 ft-lb of Torque with a Nominal Power of 5 Horsepower. The Motor is Directly Coupled to the Reel With No Gear Reduction.

Rotor Position Feedback is Provided by 3 Hall Sensors Located on the Motor Shaft. This Feedback is Used to Direct the Correct Energizing of the Motor Windings to Achieve the Desired Rotation.

Reel Assembly

Reel Motor



Reel Assembly

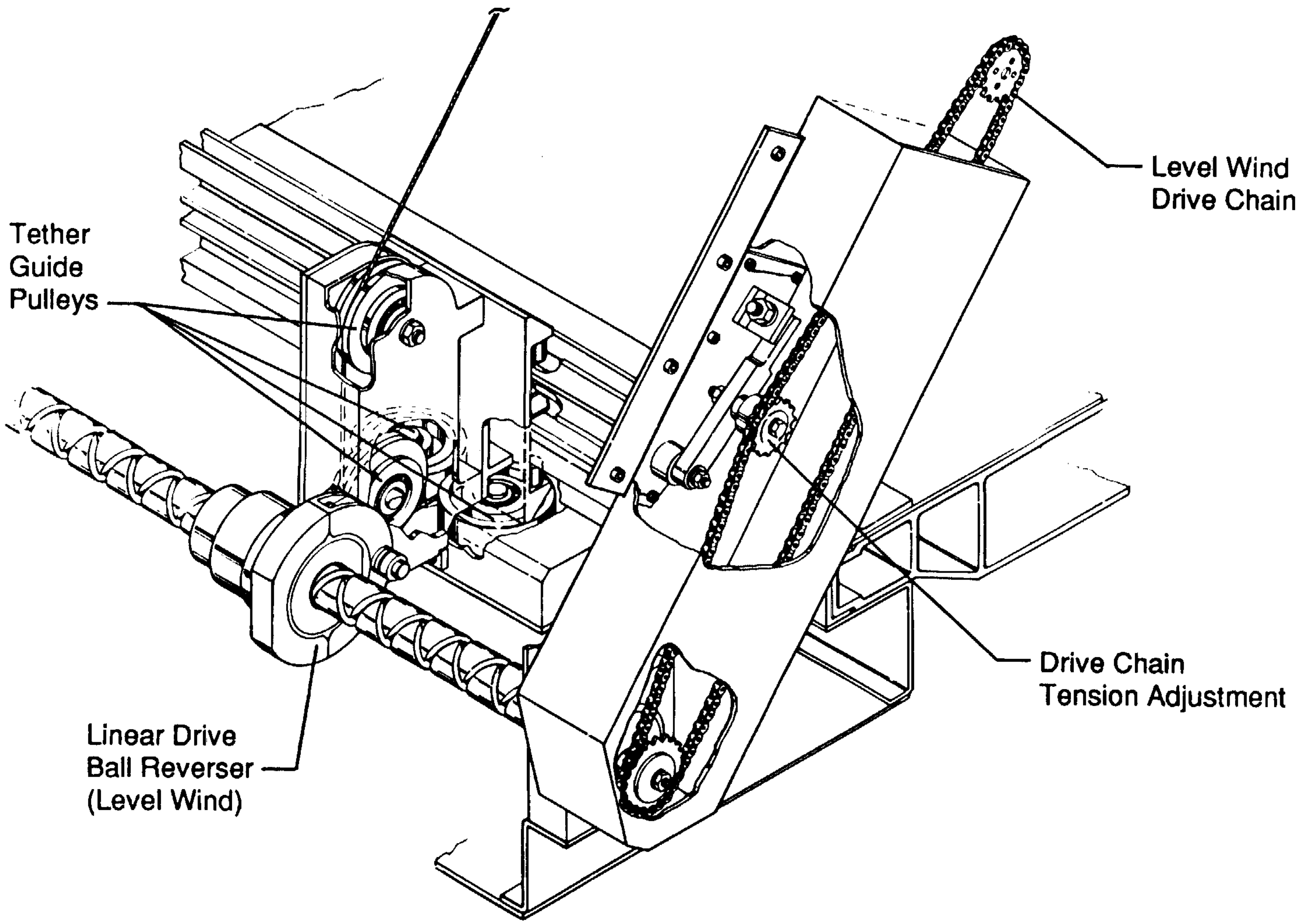
Level Wind Mechanism

The Level Wind Mechanism Uniformly Lays the Tether Across the Width of the Reel in a Compact Reversible Manner.

The Level Wind Mechanism Consists of a Chain Driven Ball Reverser Shaft, a Ball Reverser, and a Group of Tether Guide Rollers. The Level Wind Mechanism Lays the Tether Across the Reel as the Ball Reverser Traverses Back and Forth Across the Shaft. Two Horizontal Rollers Guide the Tether Coming from the LTCM and Two Vertical Rollers Guide the Tether Onto the Reel.

Reel Assembly

Level Wind Mechanism



Reel Assembly

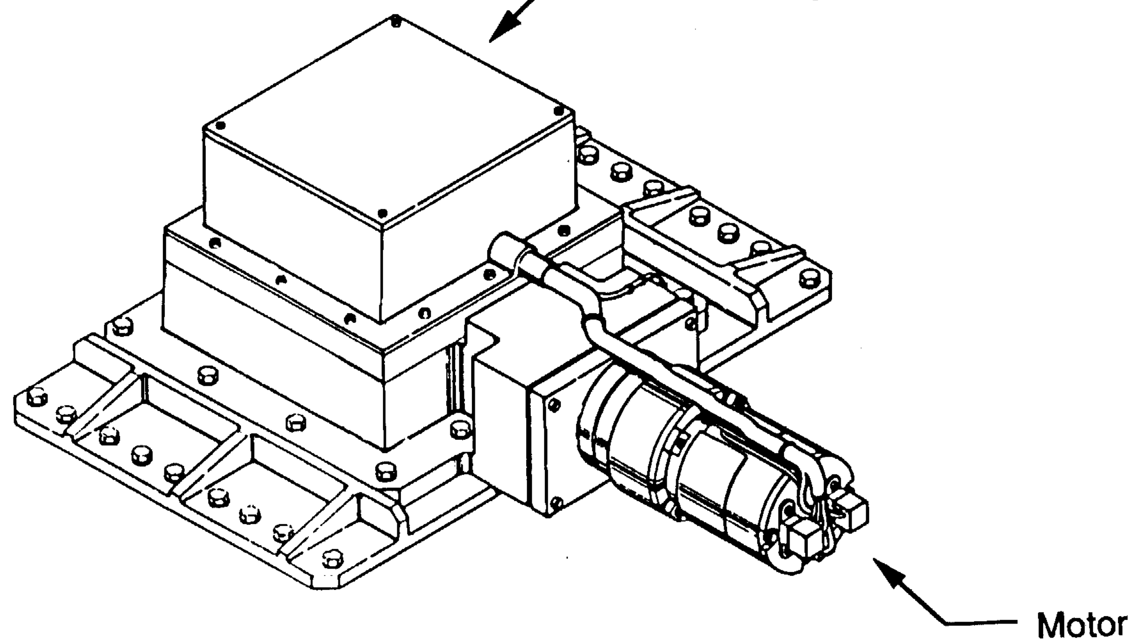
Reel Launch Lock

The Reel Launch Lock is Designed to Prevent Reel Rotation During Launch to Maintain a Preset Tether Tension. It is Released During Pre-Deployment Operations and is not Re-engaged for Landing.

The Reel Launch Lock Consists of a Gear Quadrant that Engages with Teeth Cut into the Reel Brake Rotor. The Gear Quadrant is Driven by a Nut and Leadscrew Combination. The Leadscrew is Driven by a Gear Motor. Three Spring Loaded Pins Drop into Detents in the Gear Quadrant when the Lock is Engaged. The Pins Drop Between the Rotor and the Quadrant When it is Fully Retracted, Preventing Re-Engagement. Two Micro-Switches are Activated by the Pins to Provide Verification of Fully Retracted Position.

Reel Assembly

Reel Launch Lock



Reel Assembly

Reel Brake Assembly

The Reel Brake Assembly is a Discrete On/Off Device that Provides Braking Torque to the Coupling Between the Reel and the Reel Motor. The Reel Brake Performs the Following Function:

- Stops Deployment or Retrieval of the Satellite Upon Aft Flight Deck Command.
- Operates During Boom Extension to Retain the Satellite in the Docking Ring.
- Stops Deployment or Retrieval When Pre-Set Velocity Limits are Reached.
- Stops Deployment or Retrieval Upon Loss of Power to the Brake or Tether Rate Sensors.

The Reel Brake Assembly Consists of a Spring Actuated Disc Brake Assembly, an Electromagnetic Motor, and an Electromagnetic Brake.

The Brake is Applied by the Force from 2 Parallel Springs Which Force 2 Caliper Arms Apart, Rotating the Arms About a Pivot and Forcing the Brake Pads on the Rotor.

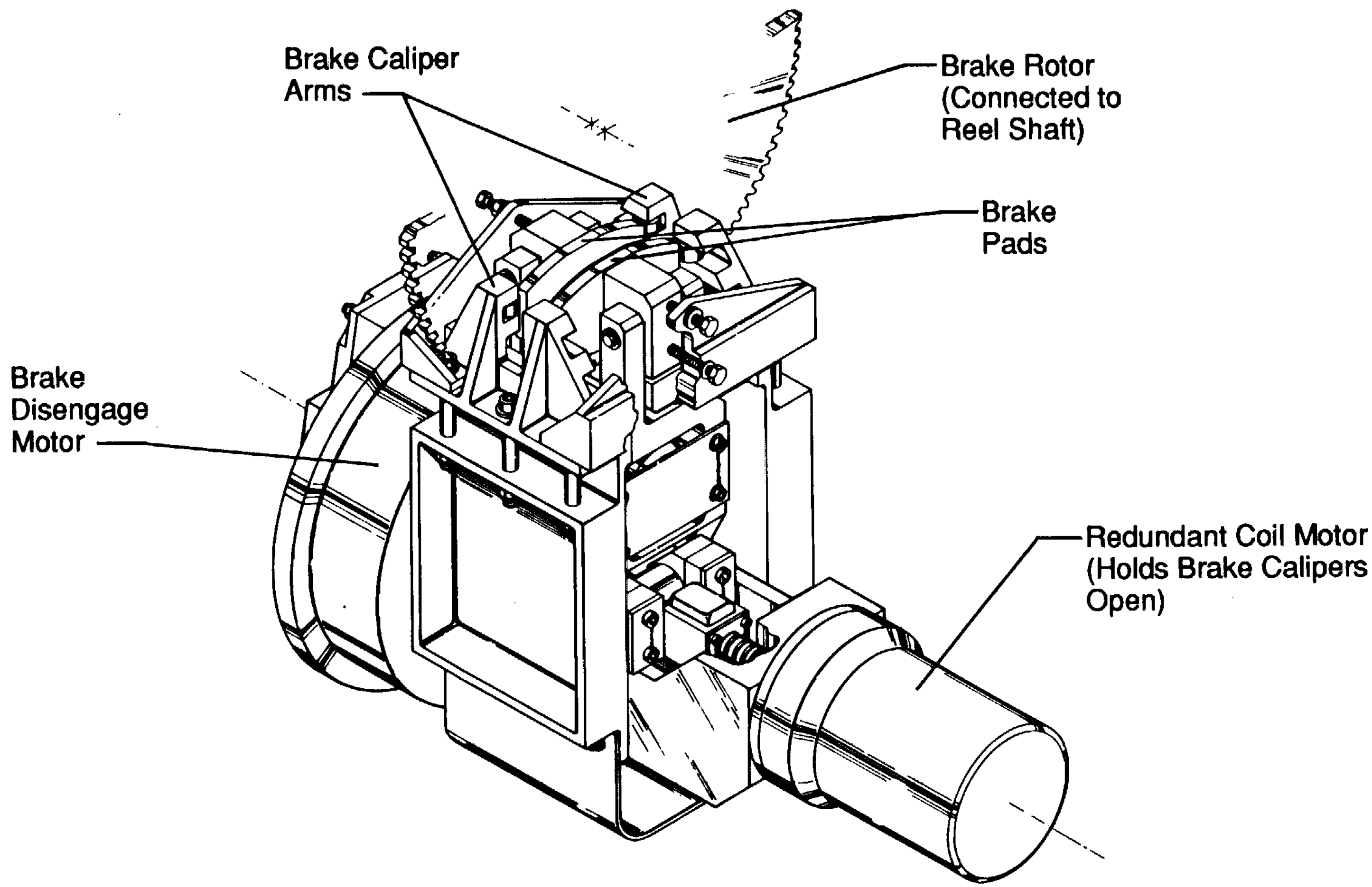
The Brake is Released by Applying Power to the Electromagnetic Motor Which Overcomes the Spring Motor Torque and Rotates a Ball Screw, Which Presses the Caliper Arms Together, Releasing the Brake.

The Electromagnetic Brake Holds the Ball Screw in the Brake-Off Position. When Power is Removed from the Electromagnetic Brake, the Spring Motor Rotates the Ball Screw in the Direction to Open the Caliper Arms and Apply the Brake.

Redundant Micro-Switches Detect the Caliper Arm Position to Provide Position Feedback to the Electromagnetic Motor, Electromagnetic Brake, and the Deployer Motor Control Assembly.

Reel Assembly

Reel Brake Assembly



Slip Ring Assembly

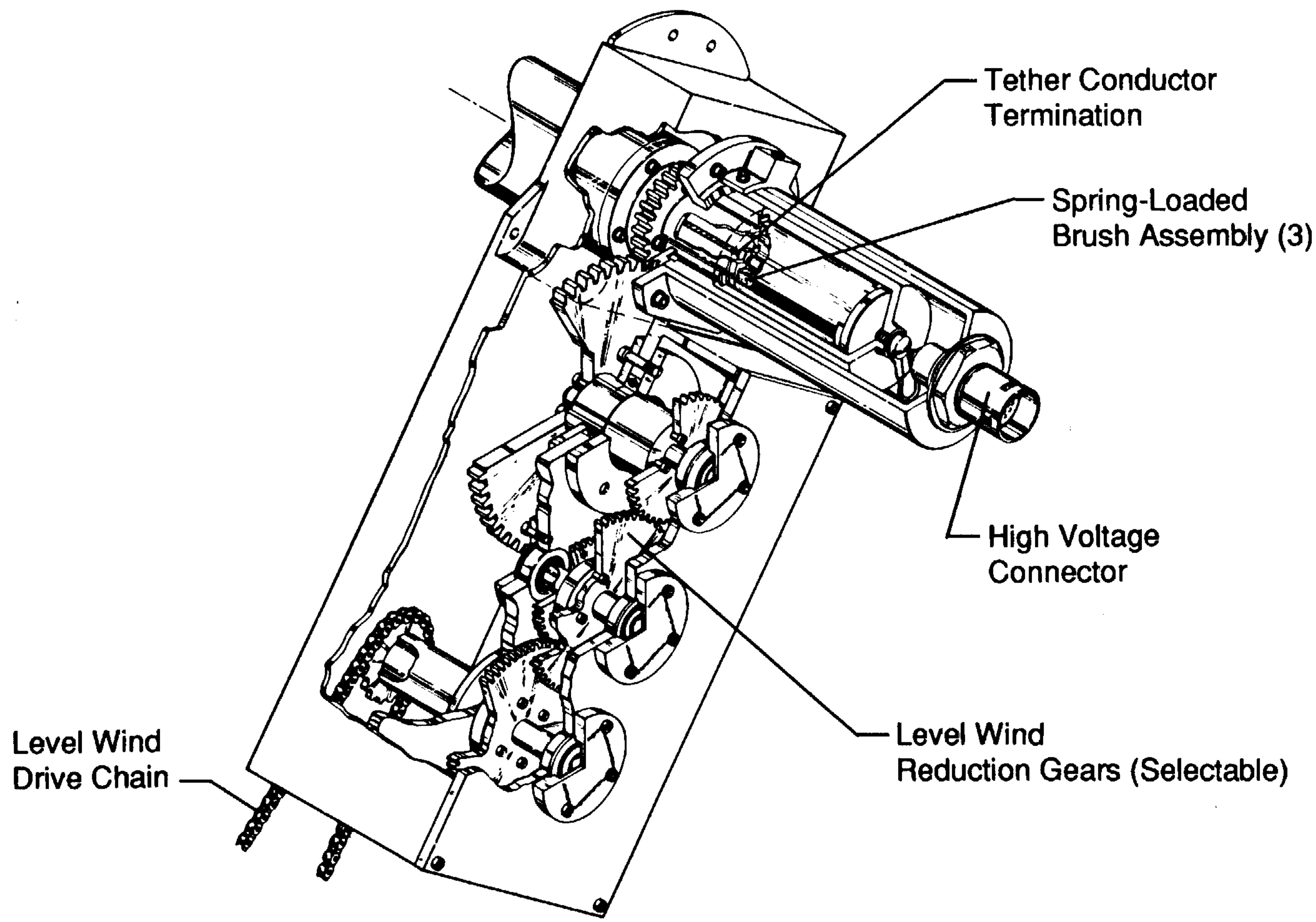
The Slip Ring Assembly Routes the Tether Current from the Tether, Which is Terminated at the Slip Ring Assembly, to the Science Instruments Via the Deployer Master Switch.

The Slip Ring Assembly Consists of the Tether Termination Device, the Slip Ring, Brushes, and Cabling.

The Slip Ring Assembly is Capable of Conducting a Continuous Current of 0.5 A and a Peak Current of 2.5 A for 3 minutes, at a Maximum Potential of 10 kVdc.

Reel Assembly

Slip Ring Assembly



Reel Assembly

Tether

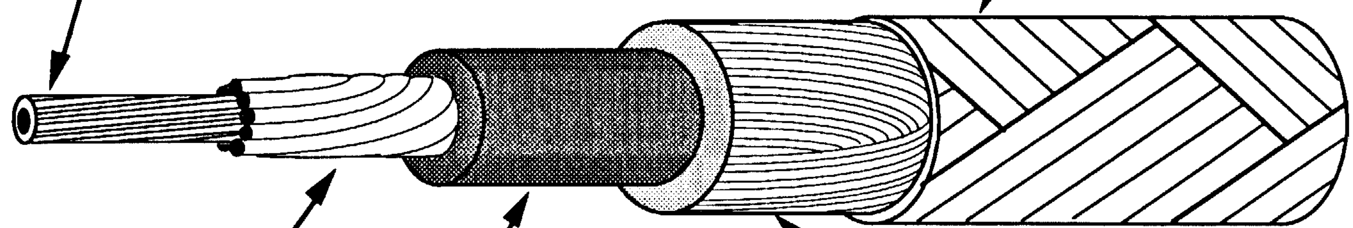
The Tether for the TSS-1 Electrodynamic Mission is a 22 Km Electrically Conducting Tether Which Provides a Mechanical and Electrical Connection Between the Deployer and the Satellite.

The Tether Contains 4 Major Elements:

- A Twisted Copper Conductor Consisting of 10 Individual 34 AWG Wires wrapped around a Nomex™ Core.
- A Fluorinated Ethylene-Propylene (FEP) Sleeve Over the Conductor that Provides Insulation Up to 10 kV.
- A Kevlar™ (Aramid™ Fiber) Strength Member Sleeve Over the Insulated Copper Wire.
- A Nomex™ Sleeve Over the Strength Member to Protect the Tether from Mechanism Induced Friction and Atomic Oxygen Degradation During Deployed Phases.

Reel Assembly

Tether



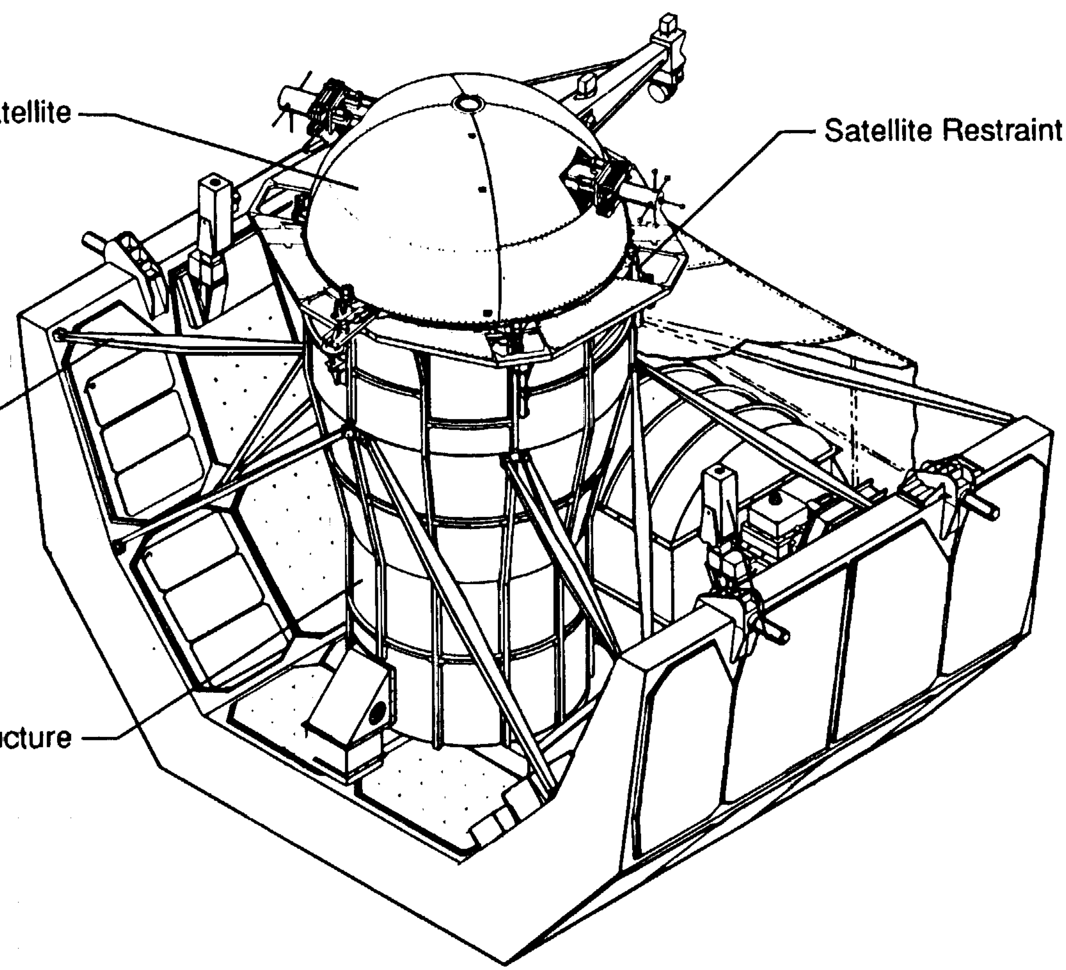
Diameter	2.54 mm (0.1 in)
Max Mass	8.2 kg/km (.0055 lb/ft or 29.0 lb/mile)
Breakstrength	1780 N (400 lb)
Temp Range	-100°C to +125°C (-148°F to +257°F)
Max Elongation	5% @ 1780 N
Elec Breakdown Voltage	10 kV (specified), 15 kV (qual)
Elec Resistance	0.12 Ω/m (specified) 0.015 Ω/m (actual @ room temp)
Leakage Current Limit	5 mA (max) @ 10 kVdc

Satellite Support Assembly

The Satellite Support Assembly Provides the 2 Main Functions of Supporting and Restraining the Satellite During Non-Deployed Mission Phases and Housing the Satellite Deployment Boom and the Boom Jettison Components.

The Satellite Support Assembly Consists of the Satellite Support Structure, the Fine Alignment System, the Satellite Restraint Latches, the Satellite Deployment Boom, the Docking Ring, and the U1 & U2 Umbilicals.

Satellite Support Assembly



Satellite Support Assembly

Satellite Support Structure

The Satellite Support Structure Supports the Stowed Satellite, the Boom, and the Mechanisms Required to Maintain and Restrain the Satellite During Non-Deployed Operations.

The Satellite Support Structure Can be Divided into the Upper, Conical, and Lower Subassemblies.

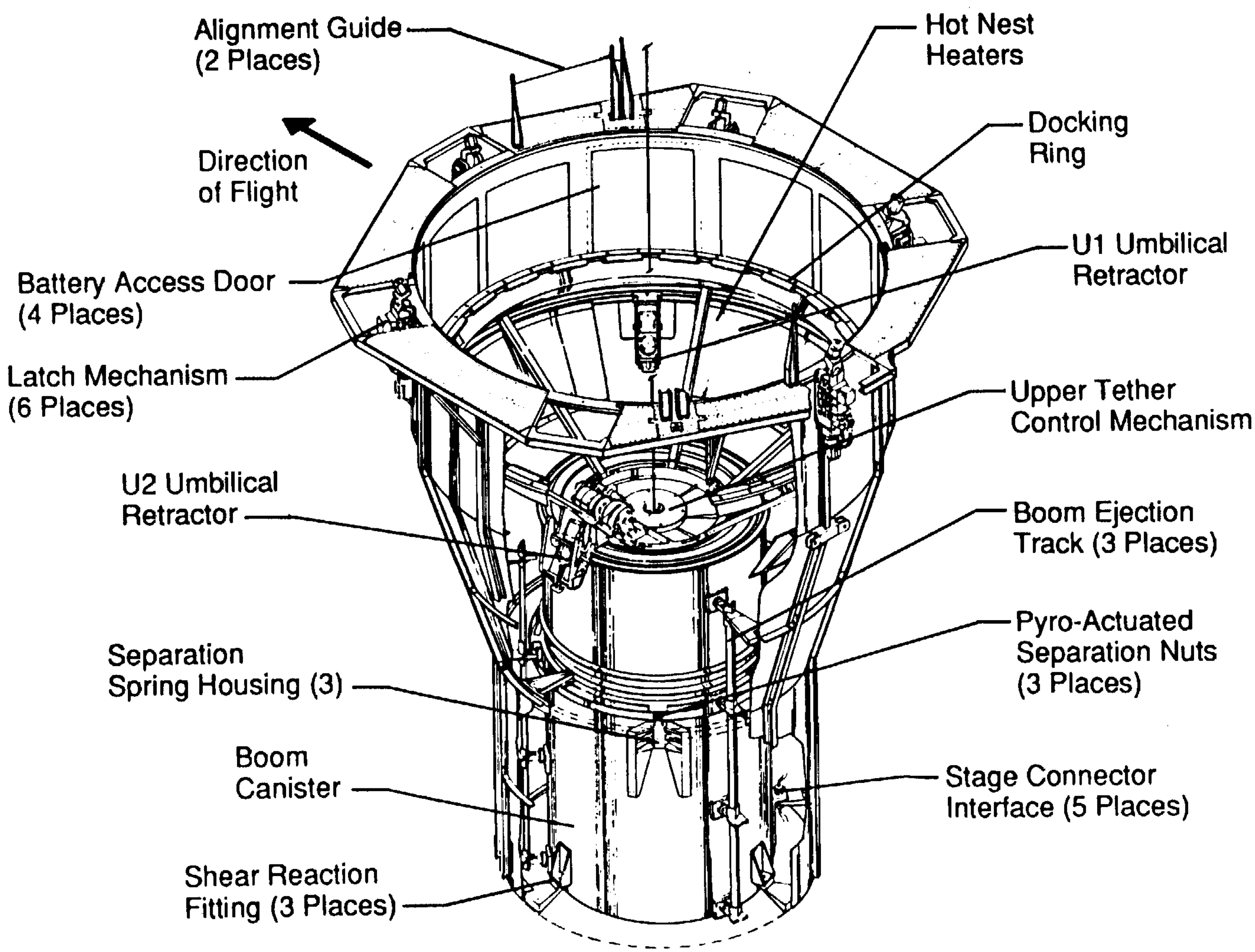
The Upper Subassembly Includes the Hexagonal 454 Ring Which Supports the Stowed Satellite and Fine Alignment System. Removal Battery Access Panels Allow Satellite Battery Installation at the Pad.

The Conical Subassembly Houses the Hot Nest Heaters which are Part of the Thermal Control System. The Truss Struts Which Attach the Satellite Support Assembly to the Pallet are Attached to the Upper Ring of the Conical Subassembly. The U1 Umbilical Mechanism is Also Mounted on the Conical Subassembly.

The Lower Subassembly is the Main Structural Interface with the Boom Canister. The Boom Jettison System Components and Staging Connectors are Located Inside the Lower Subassembly. The Lower Tether Control Mechanism is Mounted on the Exterior of the Lower Subassembly.

Satellite Support Assembly

Satellite Support Structure



Satellite Support Assembly

Fine Alignment System

The Fine Alignment System is Designed to Align the Satellite from Its Docking Attitude to Within the Capture Envelope of the Shear Wedges. It also Provides Restraint of the Satellite, Enable/Disable of the Satellite Batteries, and Electrical Grounding for the Latched Satellite.

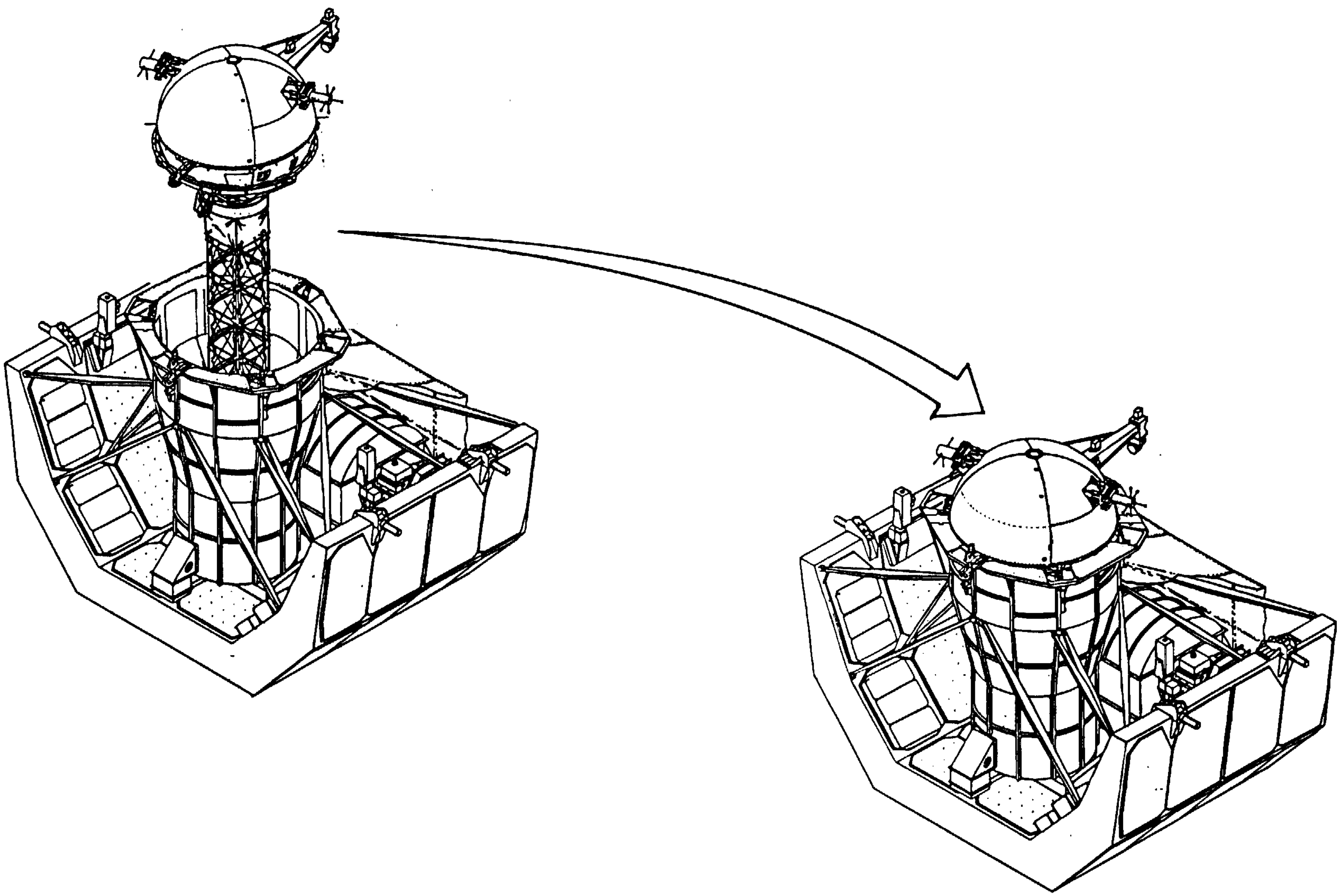
The Satellite Docking Attitude Limits are Constrained to any Yaw Angle and ±20° Roll or Pitch.

The Shear Wedge Capture Envelope Limits are ±7.62 mm (±0.300 in) (X-Y Direction), When the Satellite is Resting on the Flat Pads.

The Alignment Limits When the Satellite is on the Flat Pads is Within ±2.54 mm (±0.100 in) Along Alignment Pin Axis and ±7.62 mm (±0.300 in) Perpendicular to Alignment Pin Axis.

Satellite Support Assembly

Fine Alignment System



Satellite Support Assembly

Fine Alignment System

The Fine Alignment System Includes Both Deployer and Satellite Mounted Hardware:

2 Radially Opposed 12.7 mm (0.5 in) dia by 0.203 m (8 in) Long Alignment Pins on the Satellite.
2 Radially Opposed Alignment Bars, Towers, and Throat Plates on the Deployer.
1 Docking Ring Azimuth Drive System.

Satellite Rotation for Yaw Angle Alignment is Provided by the Docking Ring Azimuth Drive.

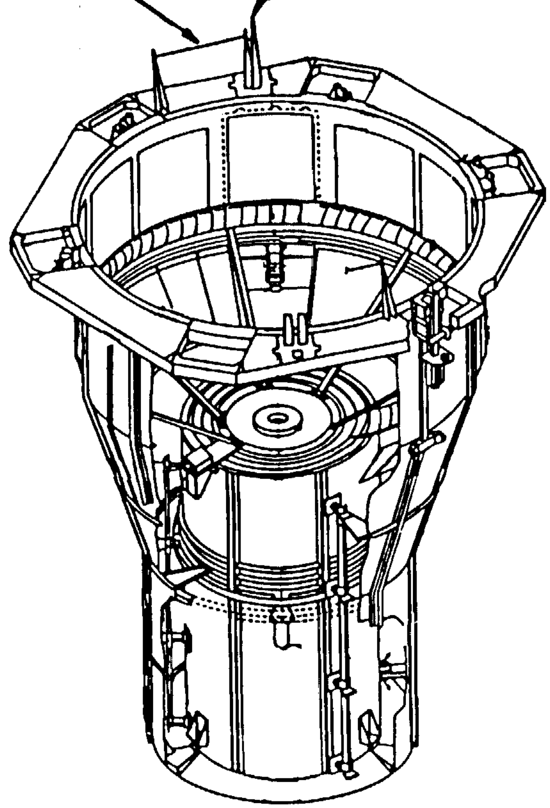
The Alignment Bars Level the Satellite When the Alignment Pins Make Contact.

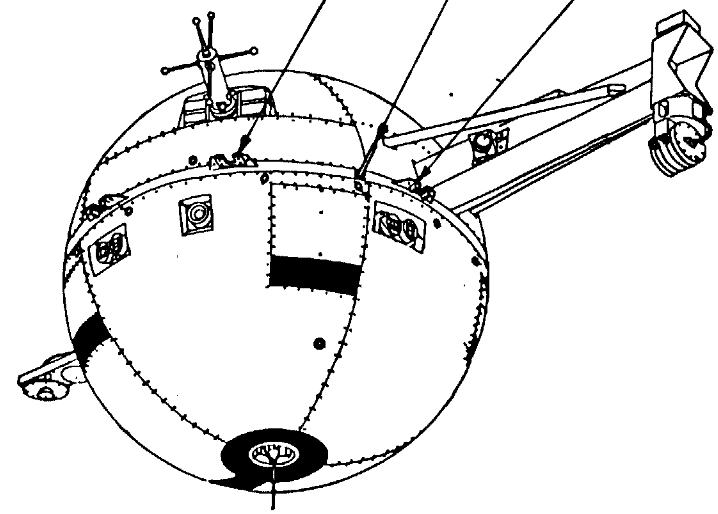
The Alignment Towers Guide the Satellite Alignment Pins into the Throat Plates which are Centered on the Alignment Pins. There is 0.762 mm (0.030 in) Clearance Between the Towers and the Pins on Each Side of the Pins.

The Unlatched Satellite Rests on Flat Pads and is Held Down by Tether Tension.

Satellite Support Assembly

Fine Alignment System





Satellite Support Assembly

Fine Alignment System

Fine Alignment Sequence:

1. The Satellite is in the Docking Ring.

2. The Docking Ring is Rotated by Crew Command to Center the Alignment Pin Over the Alignment Bar.

3. The Crew Lowers the Boom Until Both Alignment Pins Contact the Alignment Bar.

4. The Crew Retracts the Boom Until it is Fully Retracted. The Alignment Pins will be Resting on the Alignment Bars. The Satellite Will Level Itself Due to Tether Tension by Rotating Around the Alignment Pin Axis.

5. The Crew Raises the Boom to the Intermediate Stop, Which Places the Alignment Pins Above the Alignment Bars, but Below the Top of the Tall Alignment Towers.

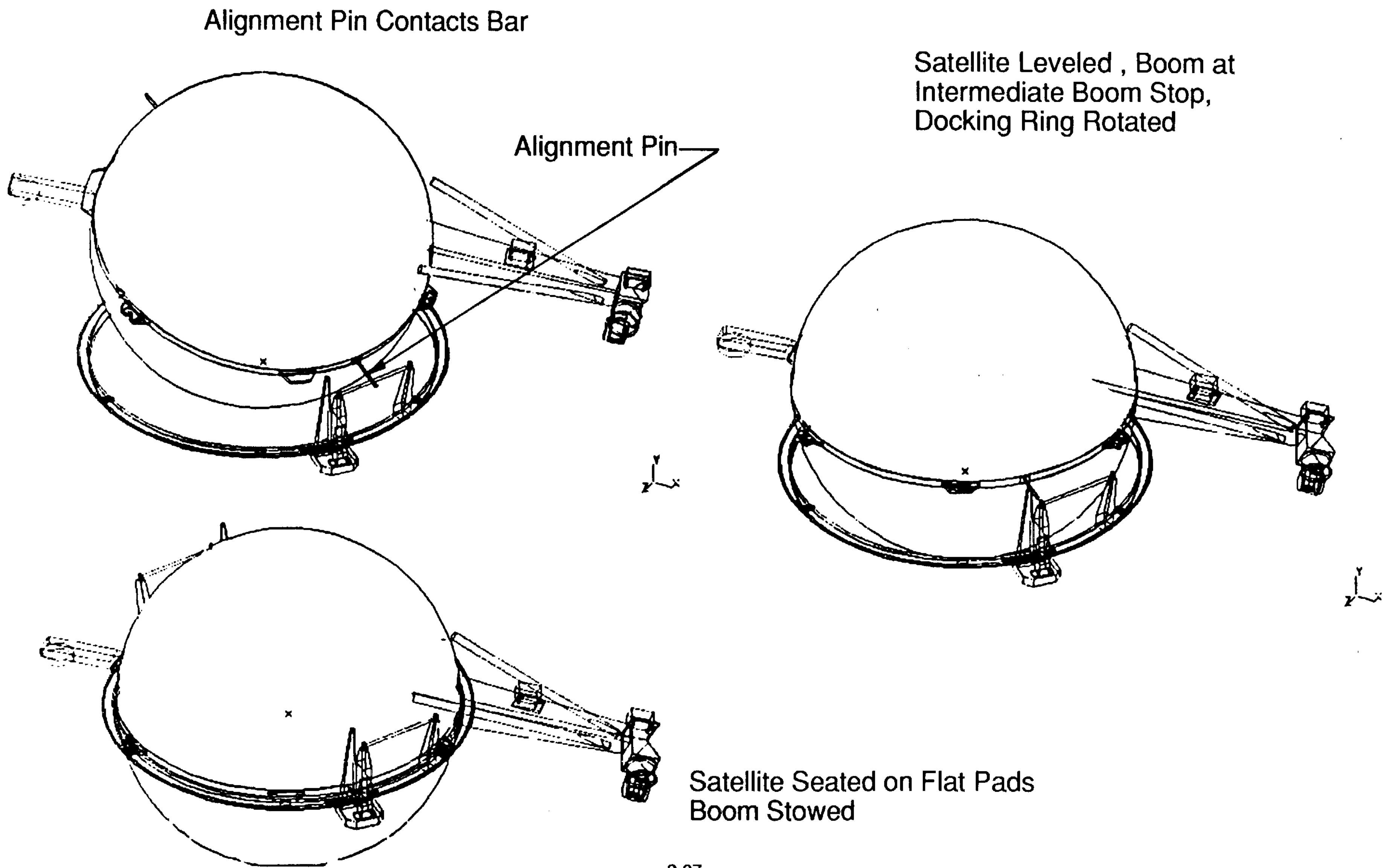
6. The Crew Rotates the Docking Ring and the Satellite Until the Alignment Pins Contact the Tall Alignment Towers and Stop Rotation of the Satellite.

7. The Crew Lowers the Boom Until it is Fully Retracted. The Satellite will be Resting on the Extended Flat Pad Plungers and Located in the X-Y Plane by the Towers and the Throat Plates.

8. The Crew Closes Latch Group 1 and Latch Group 2.

Satellite Support Assembly

Fine Alignment System



Satellite Support Assembly

Separation Switches

The Satellite Includes 4 Separation Switches which Connect the Satellite Batteries to the Satellite Electrical Power System when the Satellite is Unlatched, and Disconnect the Batteries when the Satellite is Latched. The Separation Switches Provide a Safety Backup to the Commanded Relay Which Disconnects the Batteries. The Satellite has 2 Battery Circuits, Each with 2 Parallel Separation Switches.

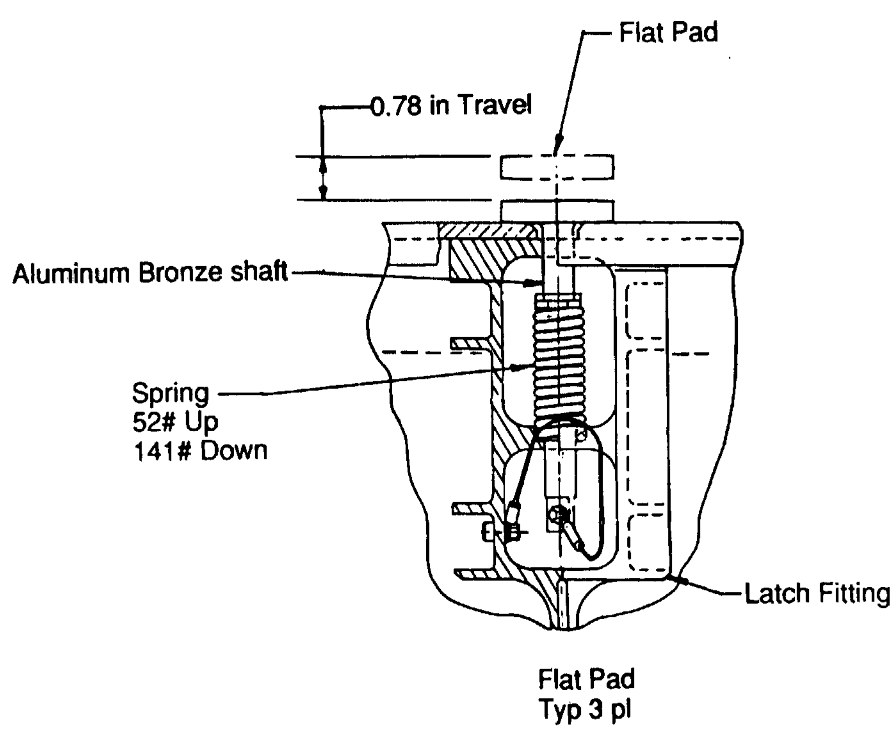
The Separation Switches are Located 90° Apart, 4° Below the Equatorial Plane. A Ball Actuator Protrudes Through the Satellite Skin. When the Ball is In, the Switch is Open; When the Ball is Out, the Switch is Closed.

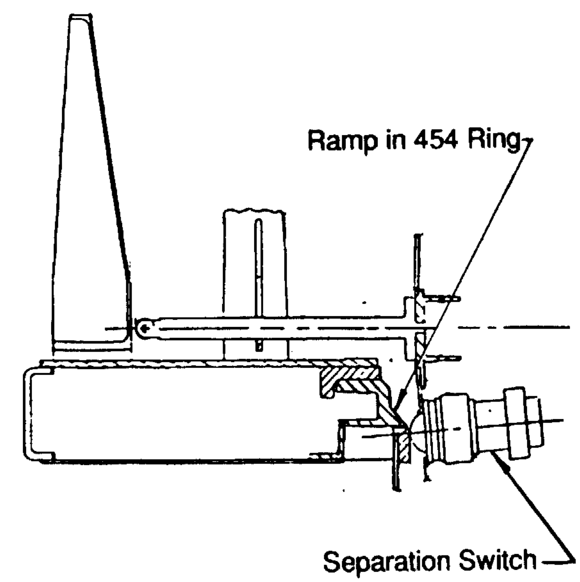
The Deployer Satellite Support Assembly Has 4 Ramps Inset into the Inside Diameter of the 454 Ring. The Separation Switch Ball Actuator is Pushed in by the Ramp as the Satellite is Latched. The Ball Actuator Rests on the Cylindrical Inside Diameter of the 454 Ring When the Satellite is Latched.

The Deployer has 3 Spring Actuated Plungers Which are Located Under the Satellite Flat Pads When the Satellite is Stowed. The Plungers Lift the Satellite 19.8 mm (0.78 in) When the Satellite is Unlatched. Latching the Satellite Forces the Plungers Down.

Satellite Support Assembly

Separation Switches





Satellite Support Assembly

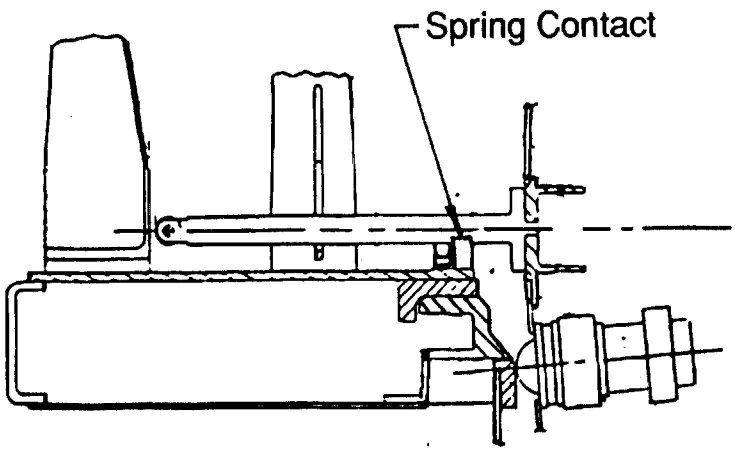
Satellite Grounding

The Satellite is Structurally Grounded to the Deployer to Provide a Return Path for Electrostatic Discharge.

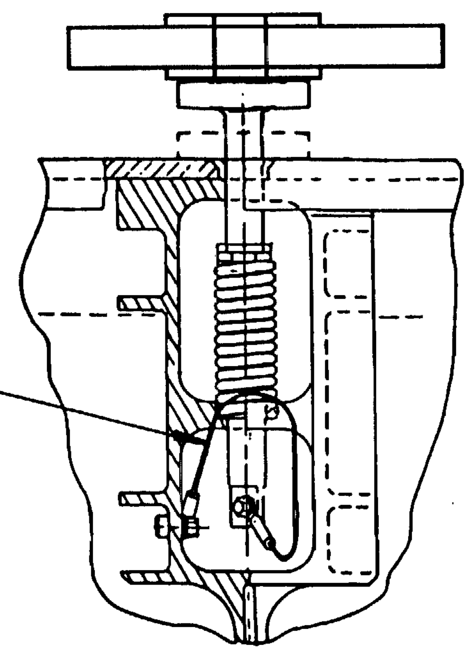
When the Satellite is Latched the Ground Path is Through the Nickel Plated Alignment Pin Which Contacts a Gold Plated Beryllium-Copper Leaf Spring Mounted on the Deployer 454 Ring. When Unlatched, the Ground Path is Through the 3 Tin Plated Satellite Flat Plates Which Rest on the Deployer's Aluminum-Bronze Flat Pads. The Flat Pads are Grounded with a Ground Strap.

Satellite Support Assembly

Satellite Grounding



Latched
Typ 2 pl



Unlatched
Typ 3 pl

Satellite Support Assembly

Satellite Restraint Latches

The Deployer has 6 Motor Driven Satellite Restraint Latches (SRL) Mounted at 60° Intervals at the top of the Satellite Support Structure to Restrain the Satellite During Non-Deployed Operations.

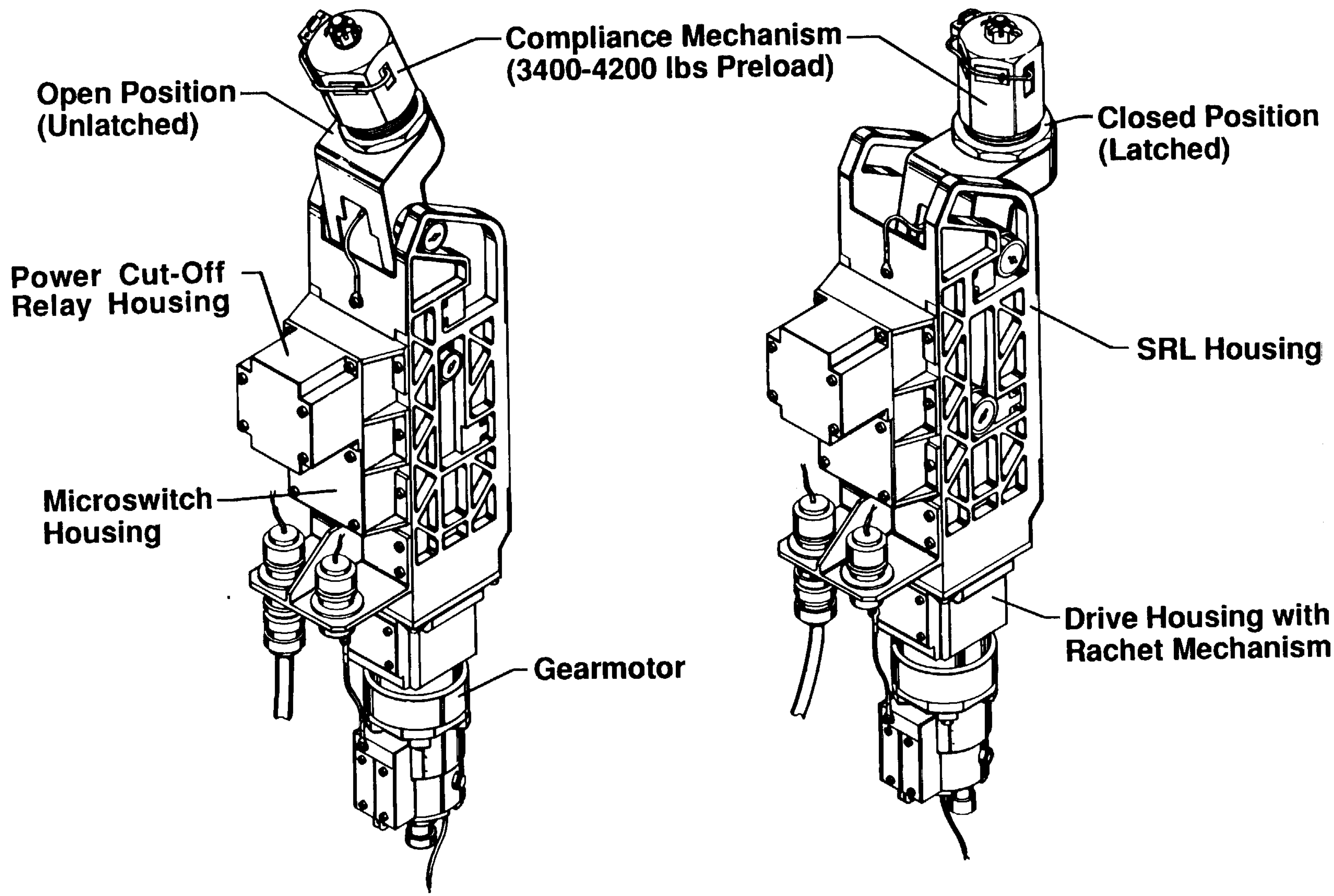
The SRLs Each Apply a Nominal 16.9 kN (3800 lb) Pre-Load to Restrain the Satellite. The Latch is Opened/Closed by Rotating a Ball Screw Through a Gearbox Which Moves the Compliance Mechanism Up or Down. A Ratchet Mechanism Prevents the SRL Preload from Backdriving the Ball Screw.

The SRLs are Commanded Open and Closed by the Crew from the Aft Flight Deck. The Electronic Drive Circuitry is Contained in the Motor Control Assembly and is Discussed in the Appropriate Section.

The SRLs Contain Limit Switches that Command a Relay to Remove Power from the Motor When the Latches reach Fully Open or Closed Positions. Additional Limit Switches Provide Feedback for Fully Open or Closed Positions.

Satellite Support Assembly

Satellite Restraint Latches



Satellite Support Assembly

Satellite Restraint Latches

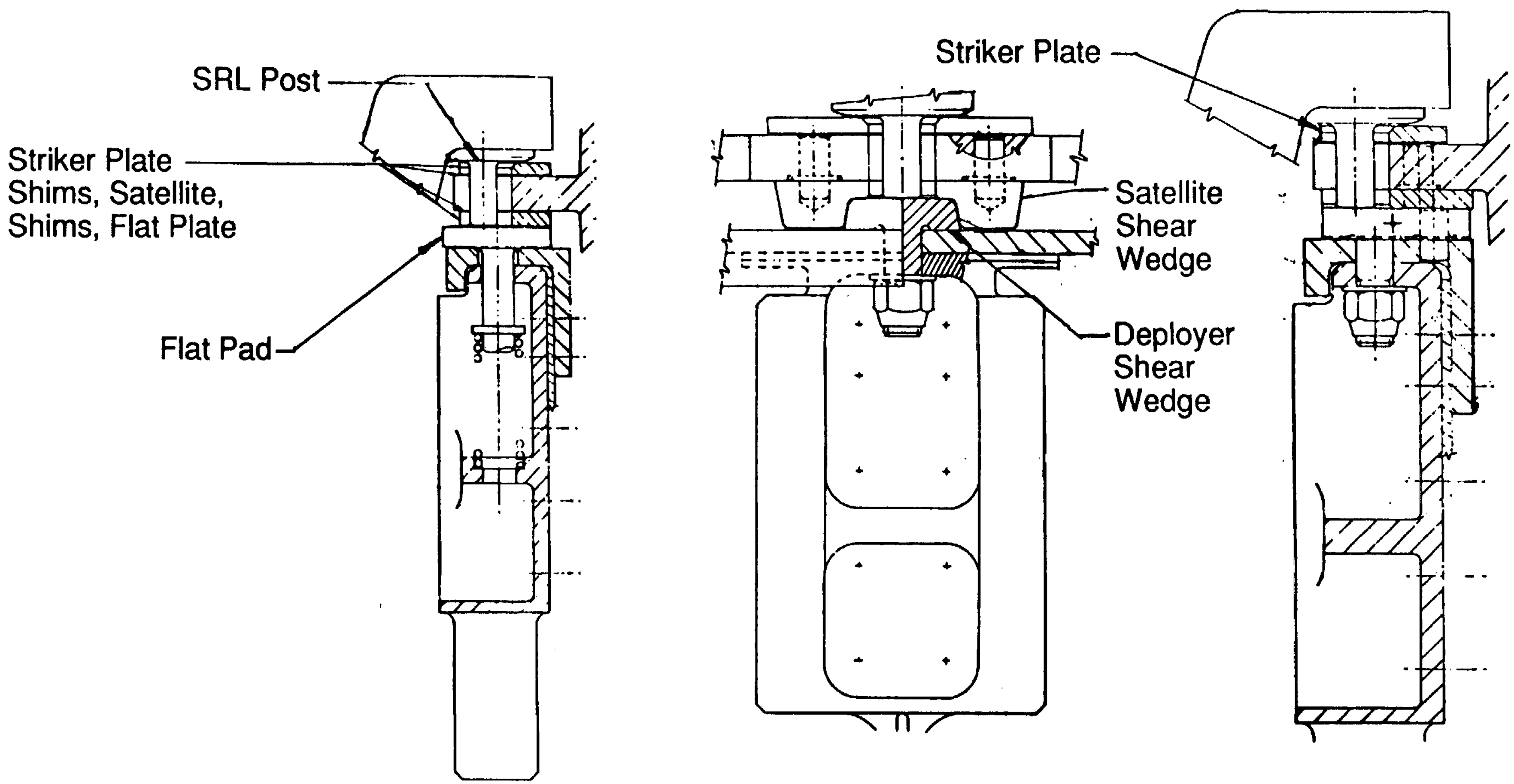
The Latched Satellite is Restrained at the 6 SRL Locations. When the Satellite is Relatched Following Deployment, the Temperature and the Size of the Satellite Will be Different than Prior to Deployment. After Relatching, the Thermal Gradients in the Satellite and the 454 Ring Will Equilibrate, Causing the Satellite to Change Shape Again.

To Prevent Unacceptably High Stresses in the Satellite, the Restraint System is Designed to Restrain the Satellite Without Clamping it.

The Satellite is Constrained in the +Z Direction by the SRL Face. The Shear Wedge/Flat Pad Provides Constraint in the -Z Direction. The Compliance Mechanism Has a Post Which is 0.010 in Longer than the Satellite Material Thickness. This Gap is Set by Shimming During Integration. The SRL Preload is Carried Through the Post Which Prevents Clamping the Satellite.

Satellite Support Assembly

Satellite Restraint Latches



Satellite Support Assembly

Satellite Restraint Latches

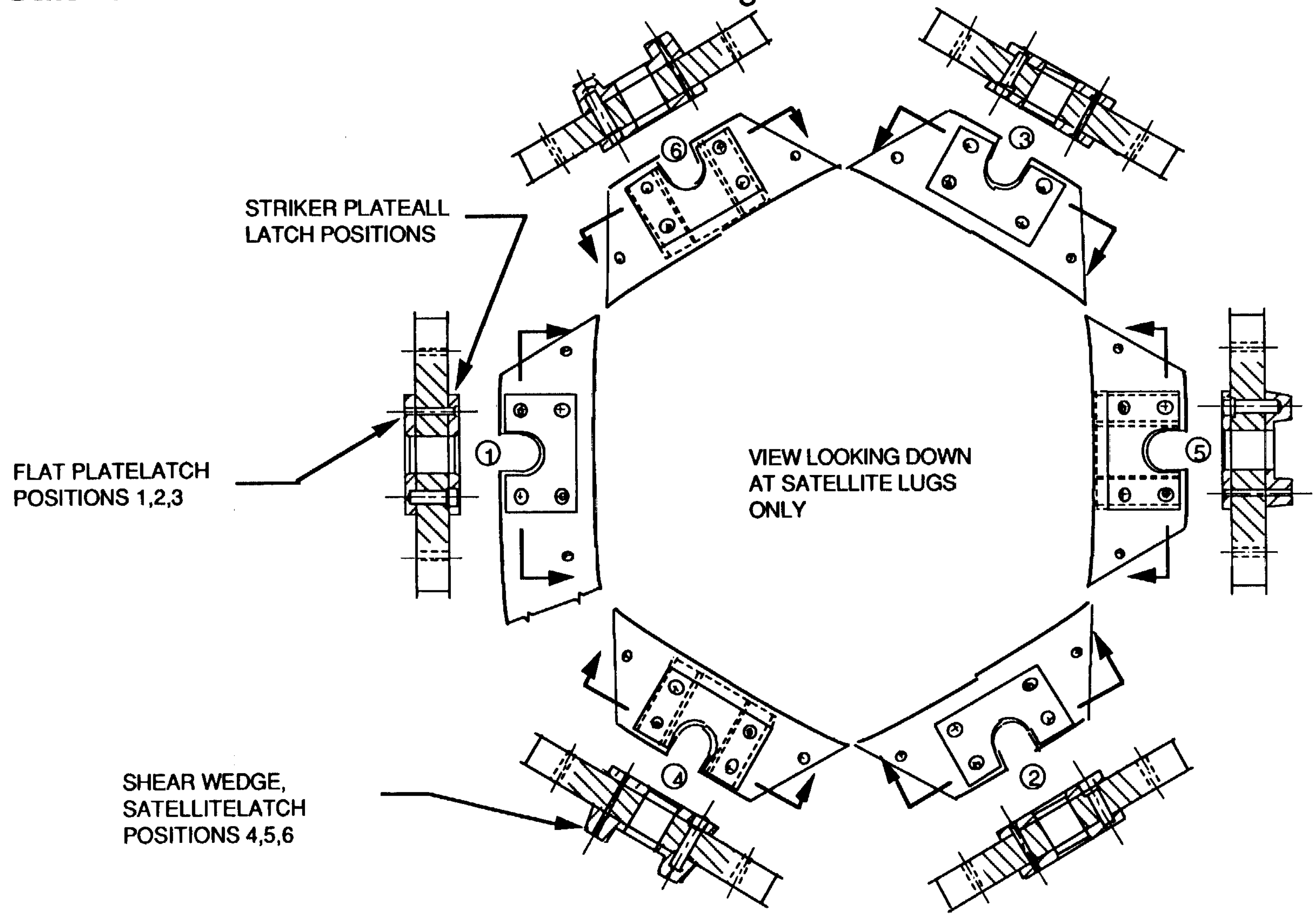
The Shear Wedges Carry all Shear Loads from the Satellite and are Located on SRLs 4,5 &6, Which are 120° Apart. The Shear Wedges are Oriented Radially to React Loads Tangentially but Remain Radially Free.

There is a 0.254 mm (0.010 in) Gap Between the Satellite and Deployer Shear Wedges to Allow Radial Movement. the Deployer Shear Wedges are Radiused to Allow Wedges to Twist Without Jamming.

The System of 3 Equally Spaced, Radially Oriented Shear Wedges Restrains the Satellite in a Unique Location, While Allowing it to Expand and Contract Radially. Tangential Distortion of the Satellite Causes the Satellite to Move in the Shear Wedges and Find a New Location.

Satellite Support Assembly

Satellite Restraint Latch Pads & Shear Wedges



Satellite Support Assembly

Satellite Deployment Boom

The Satellite Deployment Boom is an Extendable/Retractable Space Lattice Structure Designed to Position the Satellite Clear of the Orbiter's Vertical Stabilizer for Deployment and Retrieval.

General Boom Characteristics:

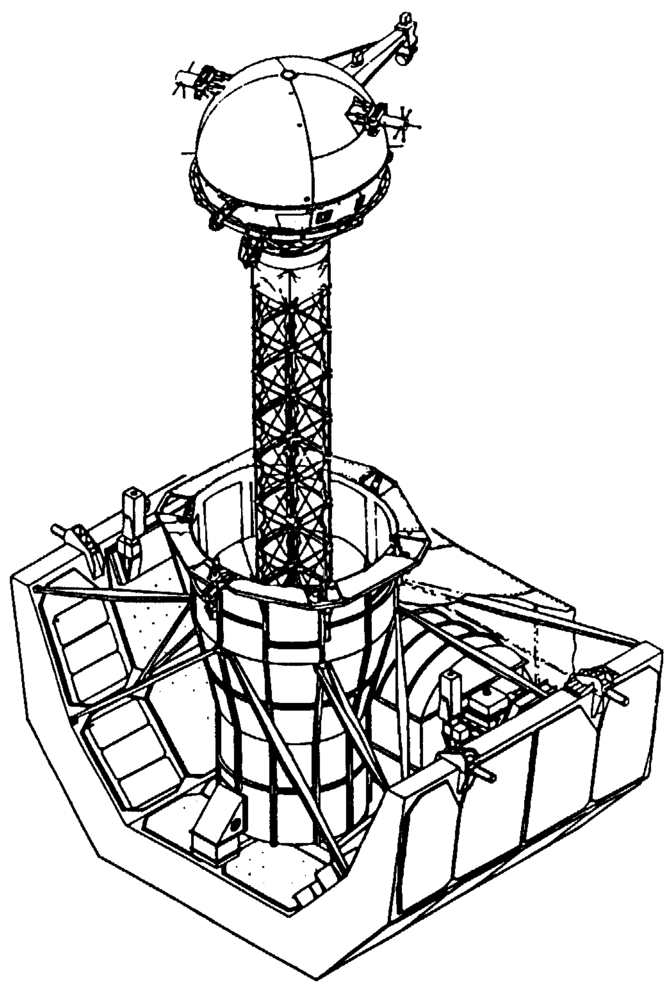
Deployed Length:	11.52 m (37' 9 3/4")
Load Capability:	
Tip Shear:	413.7 N (93 lb)
Compression:	9.63 kN (2165 lb)
Torsion:	141.8 Nm (1255 in-lb)
Deflection Spring Constant:	
Deflection < 6 in at Tip:	0.113 Nm (1 lb/in)
Deflection > 6 in at Tip:	0.565 Nm (5 lb/in)
Extend/Retract Rate:	0.777 ± 0.259 m/min (2.55 ± 0.85 ft/min)

The Satellite Deployment Boom Consists of:

- Mast
- Canister Assembly
- Electrical System

Satellite Support Assembly

Satellite Deployment Boom



Satellite Support Assembly

Satellite Deployment Boom

The Satellite Deployment Boom Mast is a Space Lattice Structure with a Square Cross-Section. The Boom Consists of 12 Individual Bays, Each 0.457 m x 0.457 m x 0.457 m (18" x 18" x 18"). The Entire Boom Folds to Store Inside the Canister. A Rotating Nut Inside the Canister Unfolds it to a Stable, Extended Configuration. The Boom Deploys Straight Out, with No Rotation.

Components:

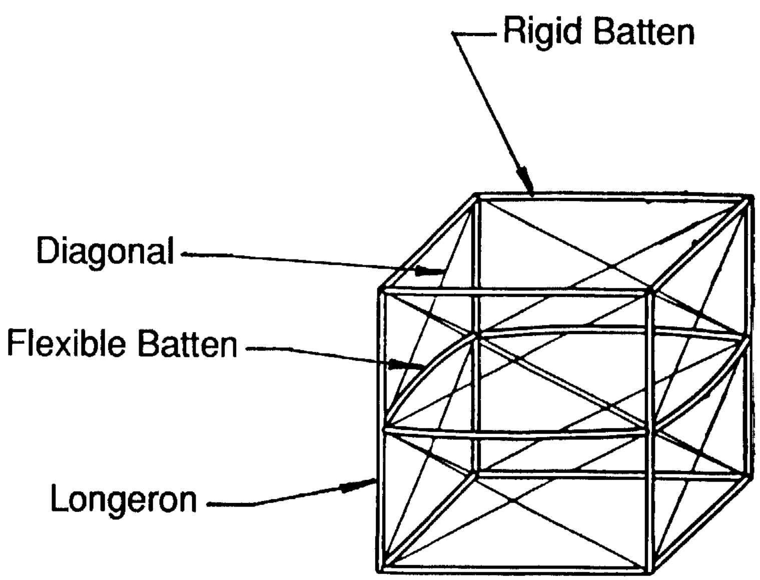
Longerons
- 4 Folding Axial Longerons Per Bay
- Solid Aluminum in the Bottom 6 Bays, Tubular Aluminum in the Top 6 Bays

Battens
- 4 Rigid and 4 Flexible Battens Per Bay
- Alternating Tubular Aluminum and Flexible Fiberglass

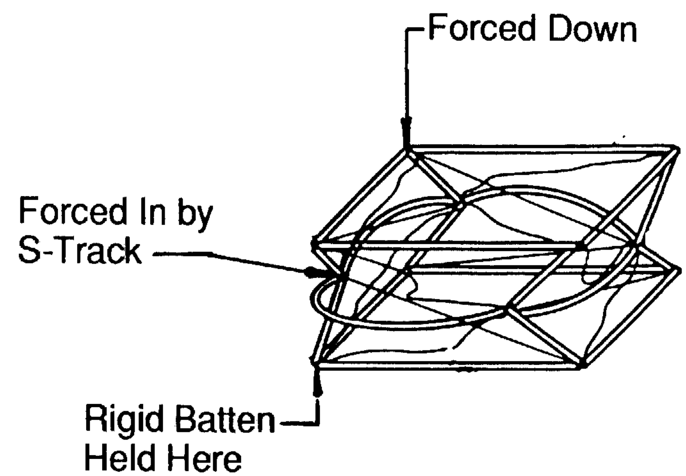
Diagonal Tensioning Cables
- Stainless Steel Cable
- 16 Per Bay

Satellite Support Assembly

Satellite Deployment Boom



Typical Bay



Stowing
Bay

Satellite Support Assembly

Satellite Deployment Boom

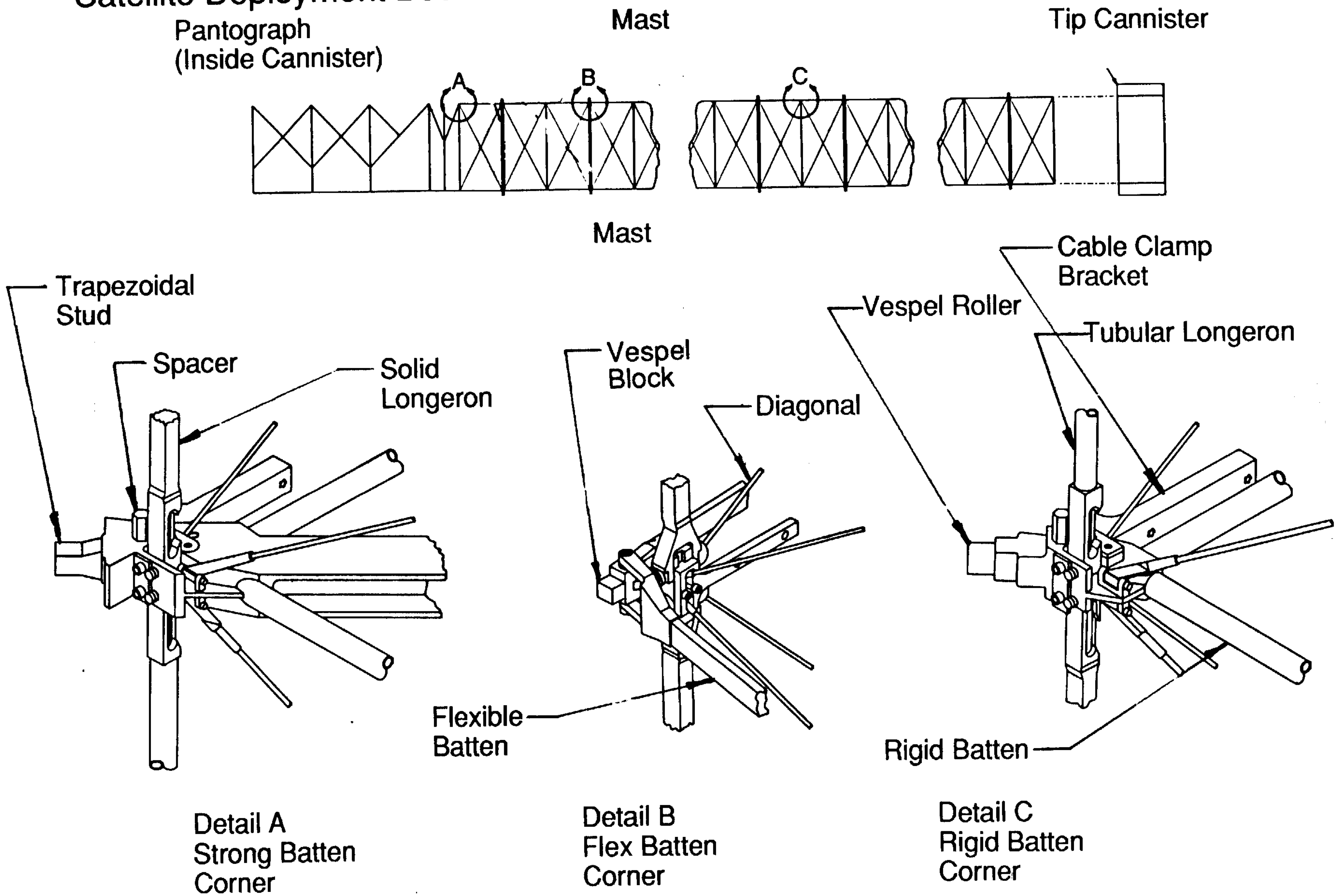
The Longerons are Folded for Storage. The Rigid Battens Form the Square End Frames for Each Bay. The Flexible Battens Hold the Bay Open When the Bay is Extended. The Diagonal Stainless Steel Tension Cables React the Flexible Batten Loads into the Rigid Battens and Define the Cubic Shape of the Deployed Bay.

The Boom Compression Load is Carried by the Longerons. Tip Shear is Carried by the Mast Truss Structure. Tip Shear is Limited by the Flexible Batten Preload. When Overloaded, the Flexible Battens Begin to Bend into the Stowed Shape and a Boom Bay(s) "Stows". The Mast Will Recover to the Extended Configuration when the Load is Removed.

A Rigid Batten and "H" Shaped Structural Diagonals are Located at the Mast Root. The Trapezoidal Stud Engages the Deployment Nut Threads When the Mast is Fully Extended. This Engagement is Required for Full Mast Strength.

Satellite Support Assembly

Satellite Deployment Boom



Satellite Support Assembly

Satellite Deployment Boom

The Canister Has a Rotating Deployment Nut with Internal Threads that Engage Rollers on the Corners of the Rigid Batten End Frames to Raise or Lower the Boom. S-Tracks on the Inside of the Canister Engage Blocks on the End of the Flexible Battens to Apply Force to the Joint, Thus Folding the Bay as it is Retracted.

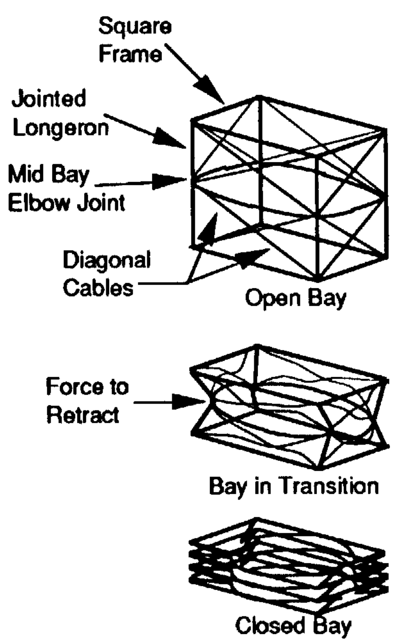
The Lower Rigid End Frame of a Bay is Held Stationary at the Bottom of the Nut While the Top End Frame Engages the Nut. As the Nut Rotates, it Forces the Upper Frame Down in the Nut While the Blocks Travel Along the S-Track and Fold the Flexible Battens. The Longerons Fold at the Flexible Batten Corners.

When the Top Rigid End Frame Reaches the Bottom of the Nut, a New End Frame is Entering the Top of the Nut. The Previous Lower End Frame is Forced Past Detents and into a Spring Loaded Frame which Holds the Collapsed Bays in the Compacted State.

Deployment Has a Similar Action. The Deployment Nut Retards the Spring Action of the Flexible Battens. The Detent Allows Only One Frame in the Nut at a Time. The Bay is Deployed as the Upper End Frame Rises Up the Nut. The Bay is Fully Deployed Before Exiting the Nut.

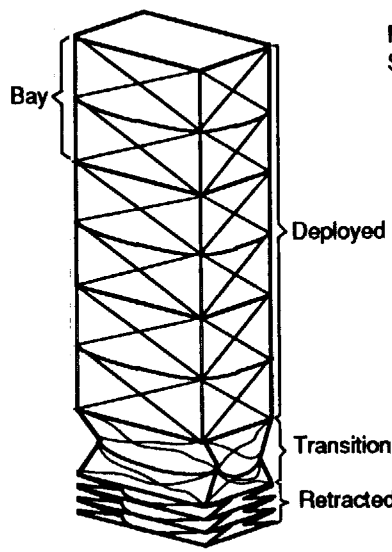
Satellite Support Assembly

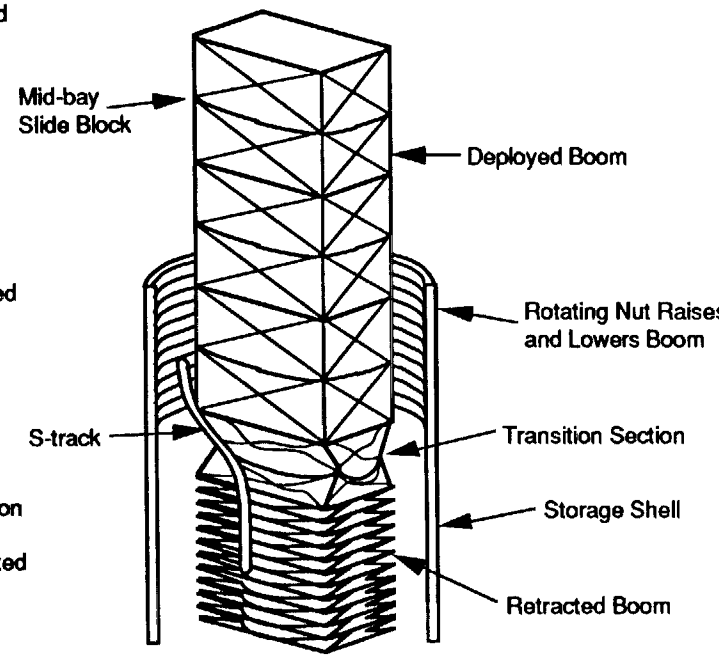
Satellite Deployment Boom



Deployable space boom consists of square aluminum end frames, folding aluminum longerons, mid-bay fiberglass bows, and diagonal tensioning cables. This ingenious construction results in a lightweight but strong boom with surprising extension capability.

Partially extended boom shows 3 fully deployed bays, 1 bay in transition, and additional bays folded flat in the retracted position.





Internal threads on large rotating nut engage rollers (not shown) to raise or lower boom.
S-track engages slide block and applies forces to mid-bay joint, thus folding bay as it is retracted.
During deployment, S-track guides transition.

Satellite Support Assembly

Satellite Deployment Boom

The Canister Houses the Stowed Mast and Supports it When Deployed. The Canister Interfaces with the Satellite Support Structure. The Deployment Mechanism is Mounted to the Outside of the Canister.

The Deployment Mechanism Utilizes Redundant Motors to Drive the Deployment Nut Through a Spur Gear Train. The Motors are Designed for Either/Or Operation with a Brake Holding the Inactive Motor. The 2 Motor Controllers are Heat Sink Mounted on the Outside of the Canister.

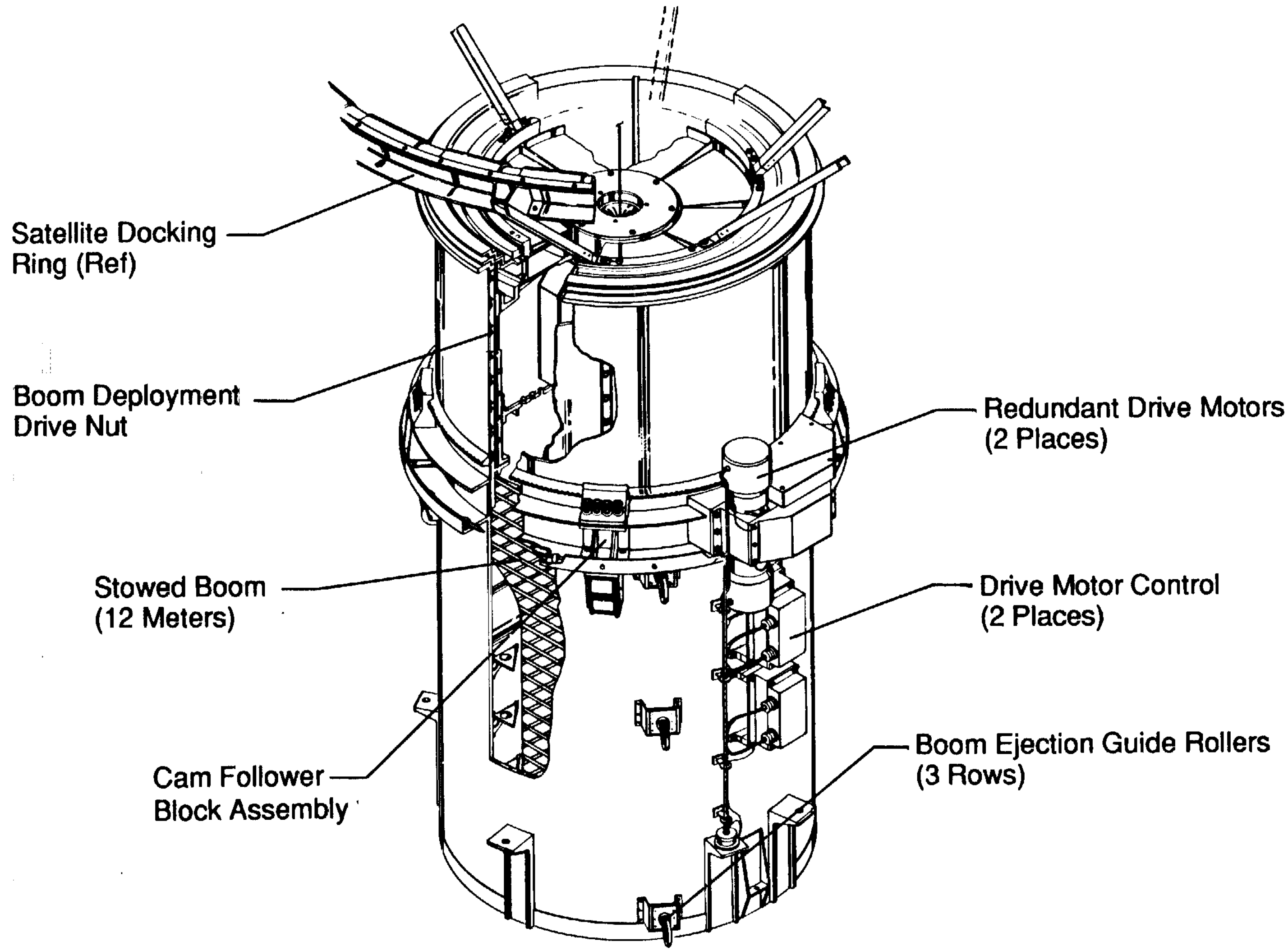
The Deployment Nut is Supported in the Canister by Cam Followers and the Ring Gear. The Cam Followers are Mounted in 4 Blocks, with 8 Cam Followers Per Block. The Blocks are Mounted on the Canister.

There are 5 Separable Connectors at the Canister Base to Join the Boom Cabling to the Deployer Cabling. The Connectors are Spring Loaded Staging Connectors Which Disconnect if the Boom is Ejected. 1 Connector Carries the Drive Motor Cabling. 4 Connectors are for the Ribbon Cable that Runs Up the Mast to the Upper Tether Control Mechanism, the Docking Ring, and the U2 Umbilical.

The Tip Canister is Attached to the End of the Mast with Screws into Corner Blocks on the Top Rigid End Frame. It Houses the Upper Tether Control Mechanism,the Docking Ring Mechanism, and the U2 Umbilical Mechanism. The Tip Canister is Stowed Inside the Deployment Nut When the Mast is Stowed. Rollers on the Tip Can Corners Engage the Deployment Nut Threads to Provide a Load Path for Launch/Landing.

Satellite Support Assembly

Satellite Deployment Boom



Satellite Support Assembly

Boom Ejection Mechanism

The Boom Ejection Mechanism is a Contingency Mode System that is Capable of Ejecting the Boom, Canister, and Docking Ring, Both with or without the Satellite Docked.

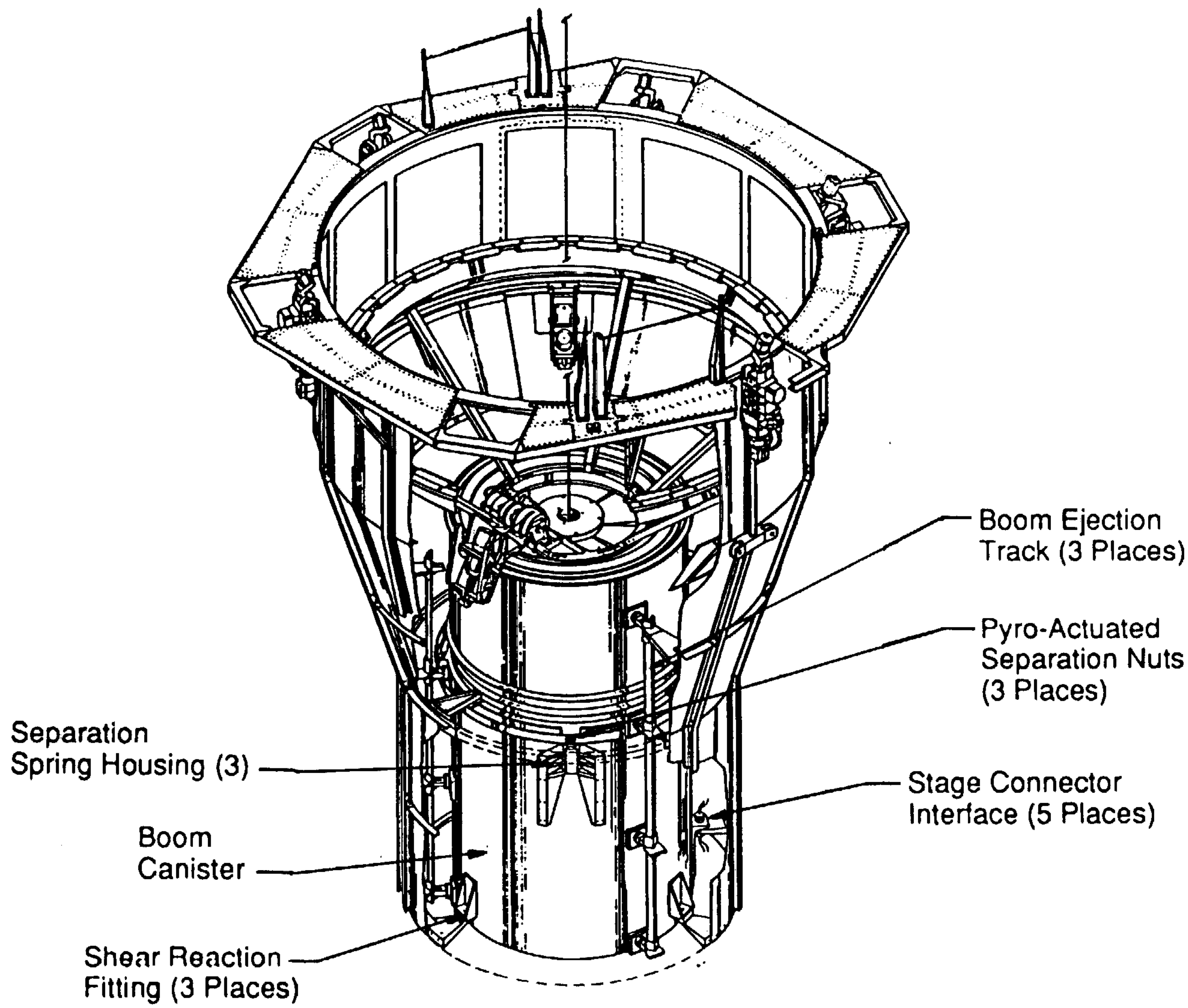
The Boom Ejection Mechanism Consists of 3 Sets of Roller Tracks, Separation Spring Assemblies, and Shear Reaction Fittings. The Separation Spring Assemblies Include a Compressed Spring that Keeps a Constant Ejection Force on the Canister. Each Spring has a Spring Constant of 102 lb/in and a Solid Height Force of 250 lbs.

The Canister is Restrained in the Satellite Support Structure by 3 Sets of Separation Nuts. Activation is by Dual NASA Standard Initiators (NSIs). When the Nuts are Activated, the Spring Force Ejects the Canister, Separating the Canister Connectors. The Canister, Boom, and Possibly the Satellite are Guided Out of the Satellite Support Structure by the Roller Tracks.

The Shear Reaction Fittings on the Bottom of the Canister are Connected to the Satellite Support Structure Through a Shear Nipple to Bear Shear Loads.

Satellite Support Assembly

Boom Ejection Mechanism



Satellite Support Assembly

Docking Ring

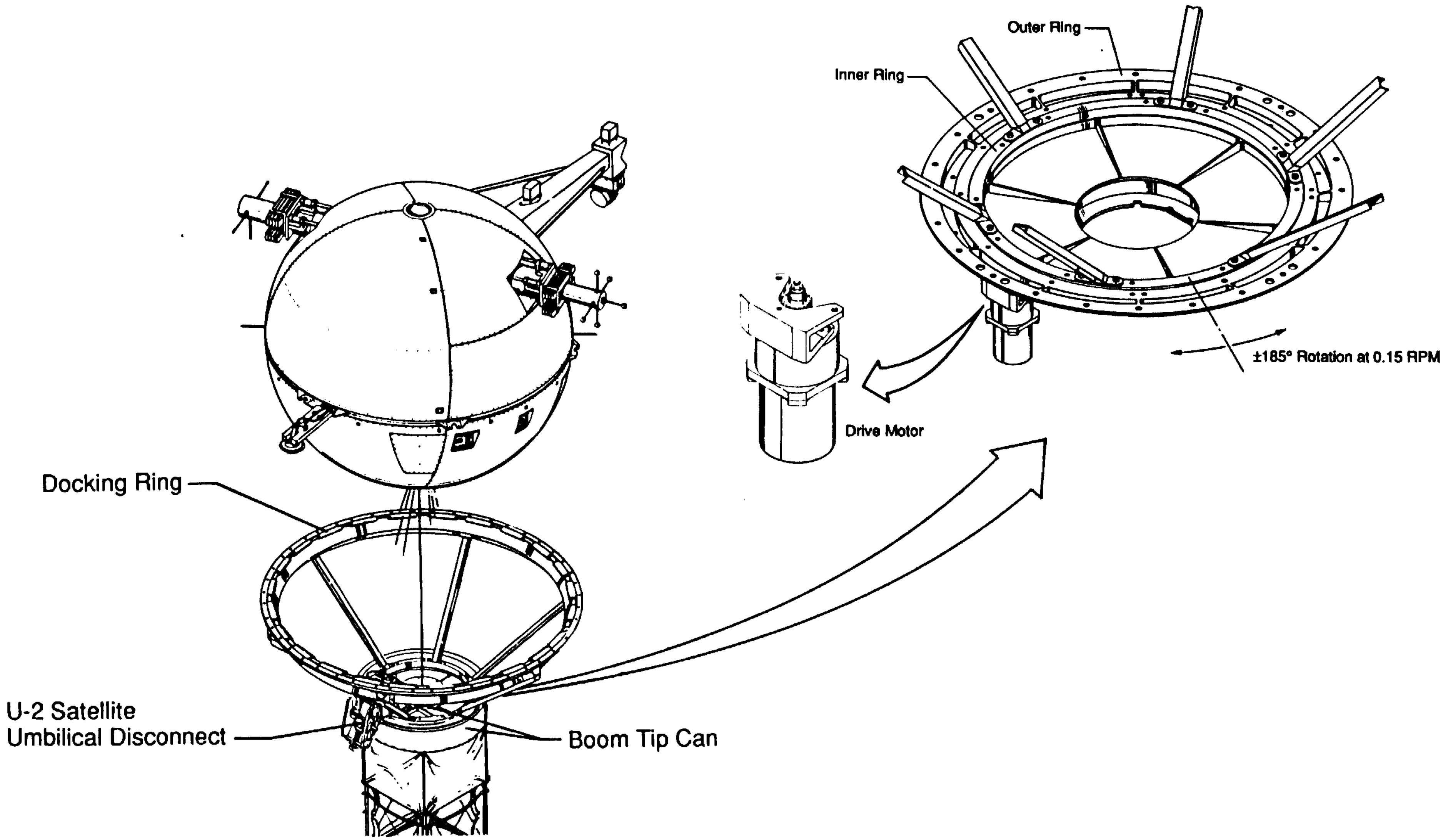
The Docking Ring is a Electromechanical Assembly Located on the Tip Canister at the End of the Boom. The Docking Ring Provides a Compliant Interface to Absorb Impact Loads when the Satellite Docks and Cradles the Satellite During Boom Extension and Retraction. It also Rotates the Satellite to Position it During Fine Alignment and to Establish the PI/RF Communication Link.

The Docking Ring Consists of a Cushioned Fiberglass Ring Attached by 6 Aluminum Struts to a Smaller Ring that Can be Rotated by the Azimuth Drive Motor.

The Docking Ring Rotates at a Rate of 0.15 ± 0.10 rpm over an Angular Range of $\pm 135°$ with the U2 Umbilical Connected and $\pm 185°$ with the U2 Disconnected. Rotation is Limited with the U2 Connected Due to Mechanical Interferences.

Satellite Support Assembly

Docking Ring



Satellite Support Assembly

U1 Umbilical

The U1 Umbilical Mechanism Utilizes a Linkage Connected to the Male Umbilical Connector. A Motor Retracts a Lanyard Cable, Producing Tension Which Disconnects the Connector. The Connector and Linkage are Retracted and Mechanically Locked.

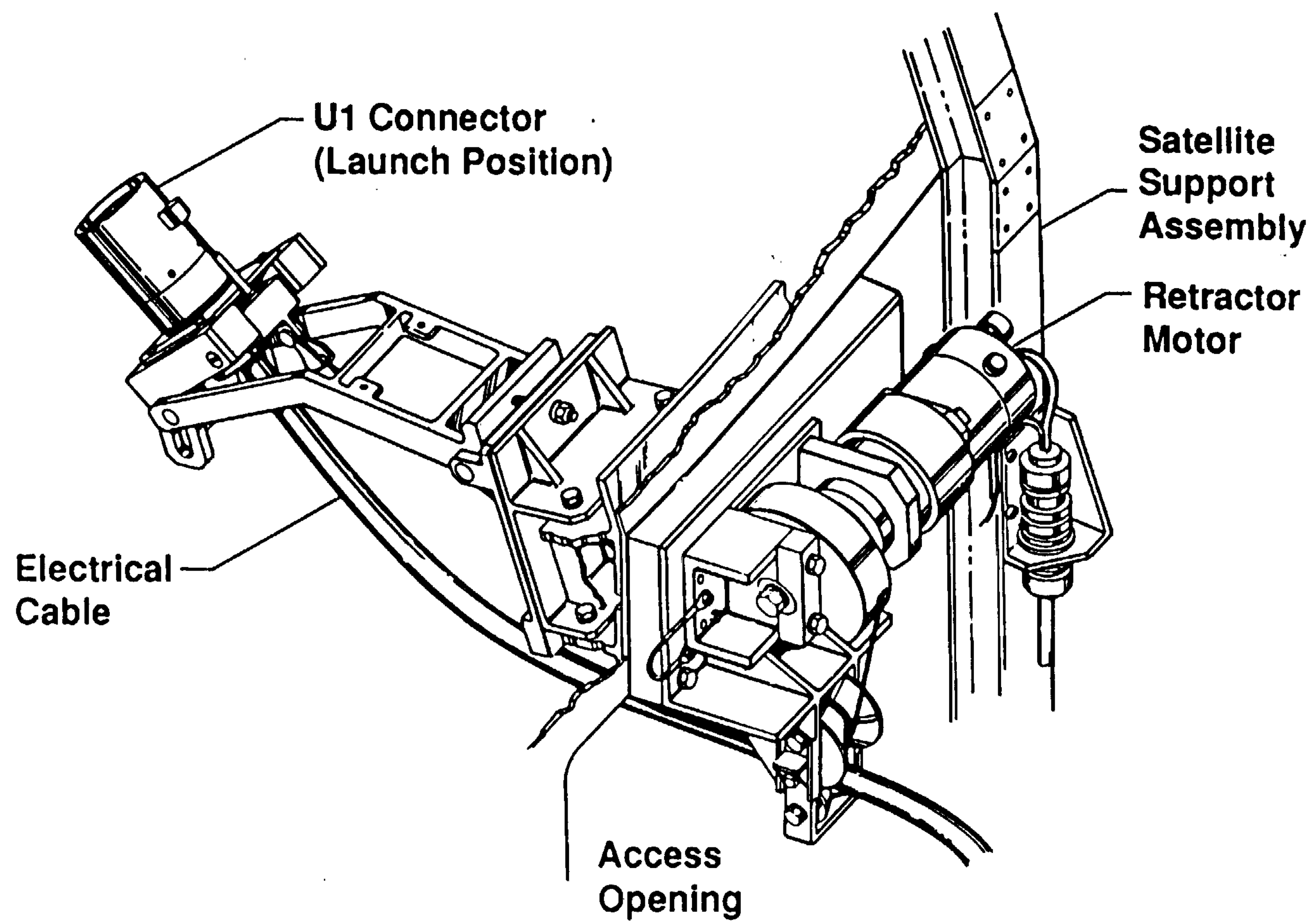
The U1 is Disconnected from the Satellite After the Satellite is Powered and Prior to Satellite Deployment Boom Extension.

U1 Disconnect is Electrically Indicated by Breaking a Turn Around Circuit. 3 Independent Microswitches Indicate that the U1 Mechanism is Fully Retracted.

The U1 Umbilical Connector Cannot be Reconnected on Orbit.

Satellite Support Assembly

U1 Umbilical



Satellite Support Assembly

U2 Umbilical

The U2 Umbilical Mechanism Utilizes a Cable and Yoke Connected to the Male Umbilical Connector. A Motor Retracts the Lanyard Cable, Producing Tension Which Disconnects the Connector. The Connector and Yoke are Retracted and Guide Pins on the Yoke Settle into Slots in the Retention Bracket. One Way Spring Clips Retain the Yoke and Connector in the Bracket.

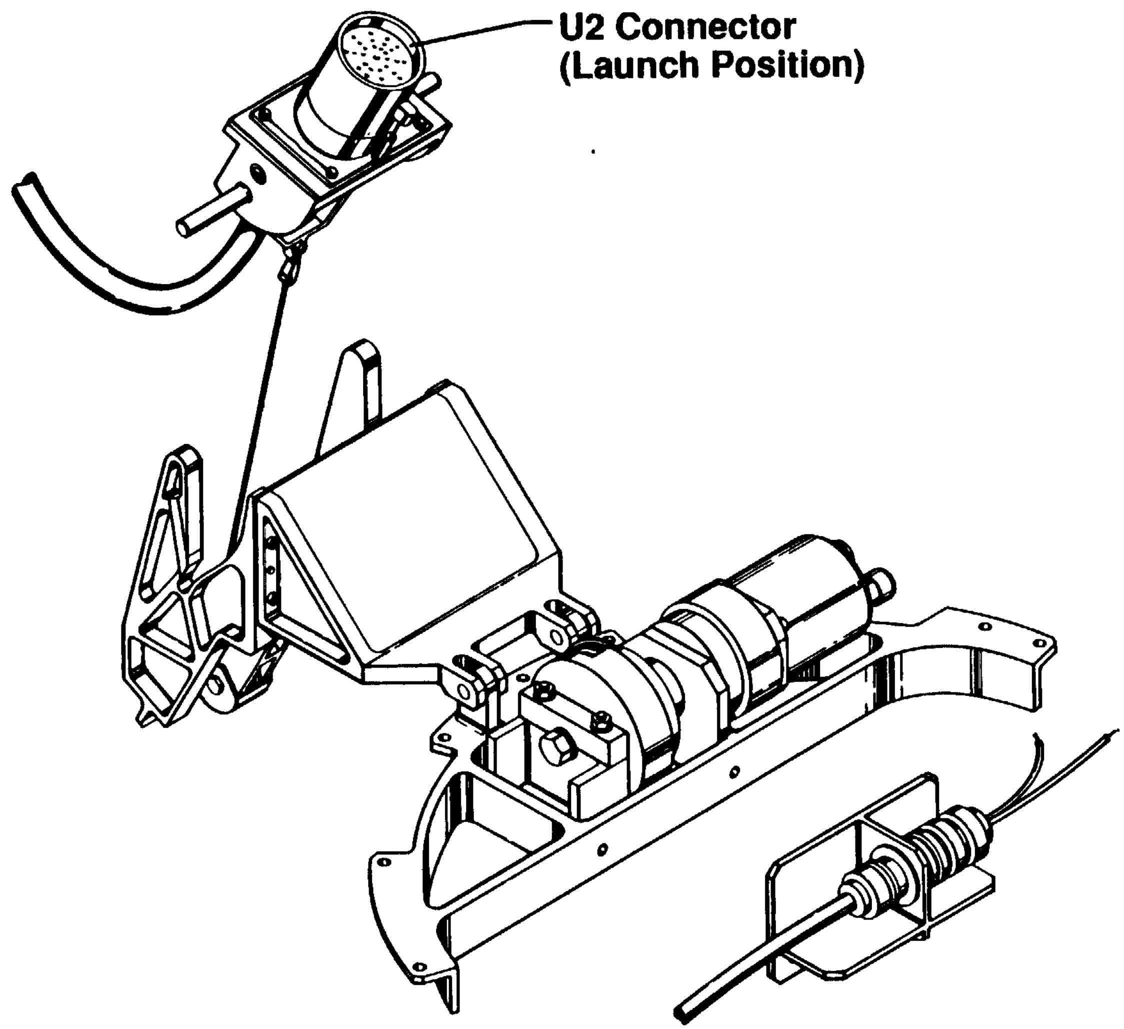
U2 is Disconnected from the Satellite After Full Boom Extension and Satellite Rotation to Establish the RF Communications Link Between the Satellite and the Orbiter.

U2 Disconnect is Electrically Indicated by Breaking a Turn Around Circuit.

The U2 Umbilical Connector Cannot be Reconnected on Orbit.

Satellite Support Assembly

U2 Umbilical



Tether Control Mechanisms

Overview

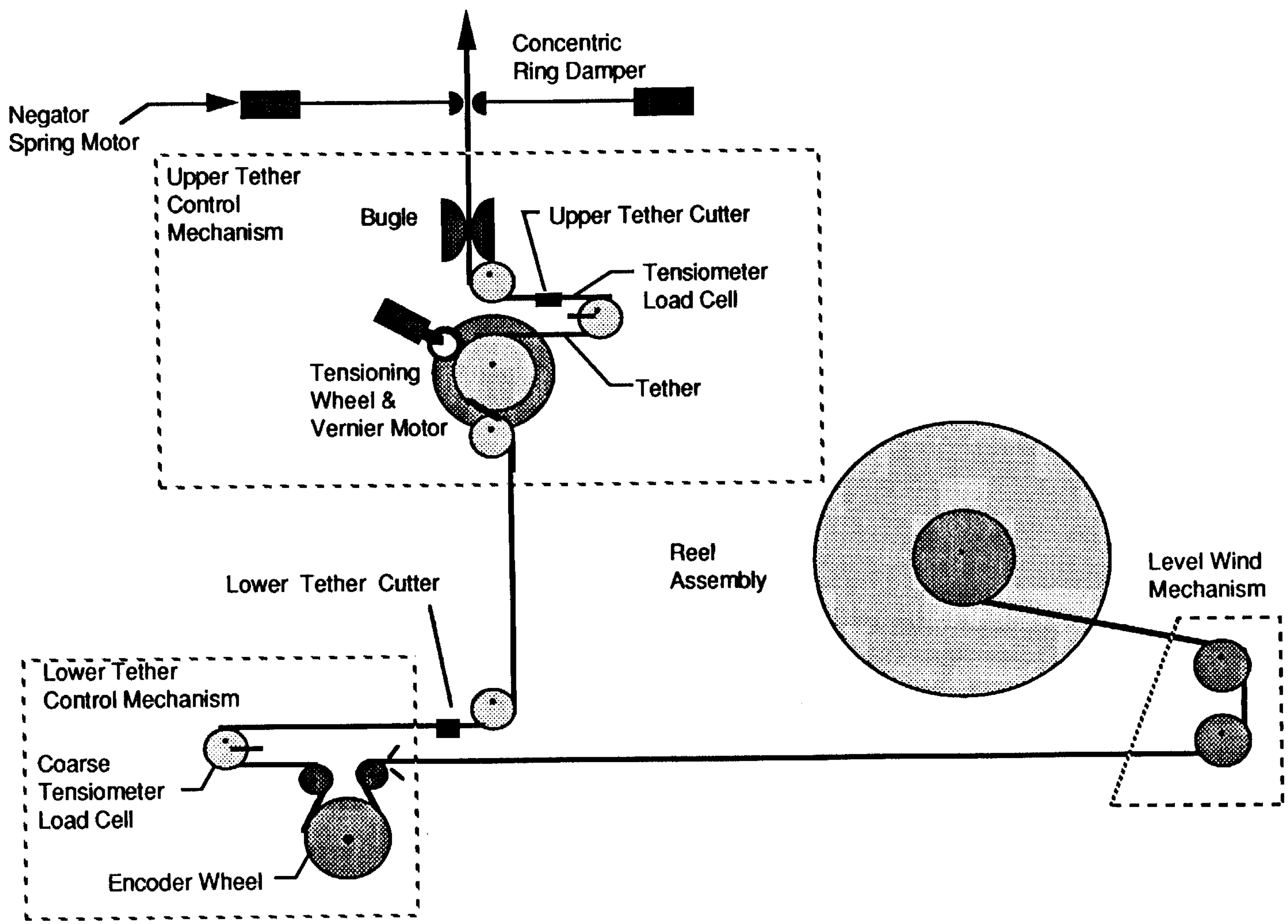
The Tether Control Mechanisms are a Group of Mechanisms that Operate in Conjunction with the Deployer Avionics to Provide Control Over the Tether During Deployment, Retrieval, and On-Station Periods of the TSS Mission.

The Tether Control System Consists of the Lower Tether Control Mechanism (LTCM), the Encoder, the Inboard Tensiometer, the Upper Tether Control Mechanism (UTCM), the Vernier Drive, the Outboard Tensiometer, the Reel Drive Mechanisms, and the Reel Brake Mechanism.

The Reel Mechanisms and the Reel Brake Mechanism Have Been Discussed in the Reel Assembly Portion of this Overview. Please Refer to that Section for the Complete Overview of these Mechanisms.

Tether Control Mechanisms

Tether Control System Components



Tether Control Mechanisms

Lower Tether Control Mechanism

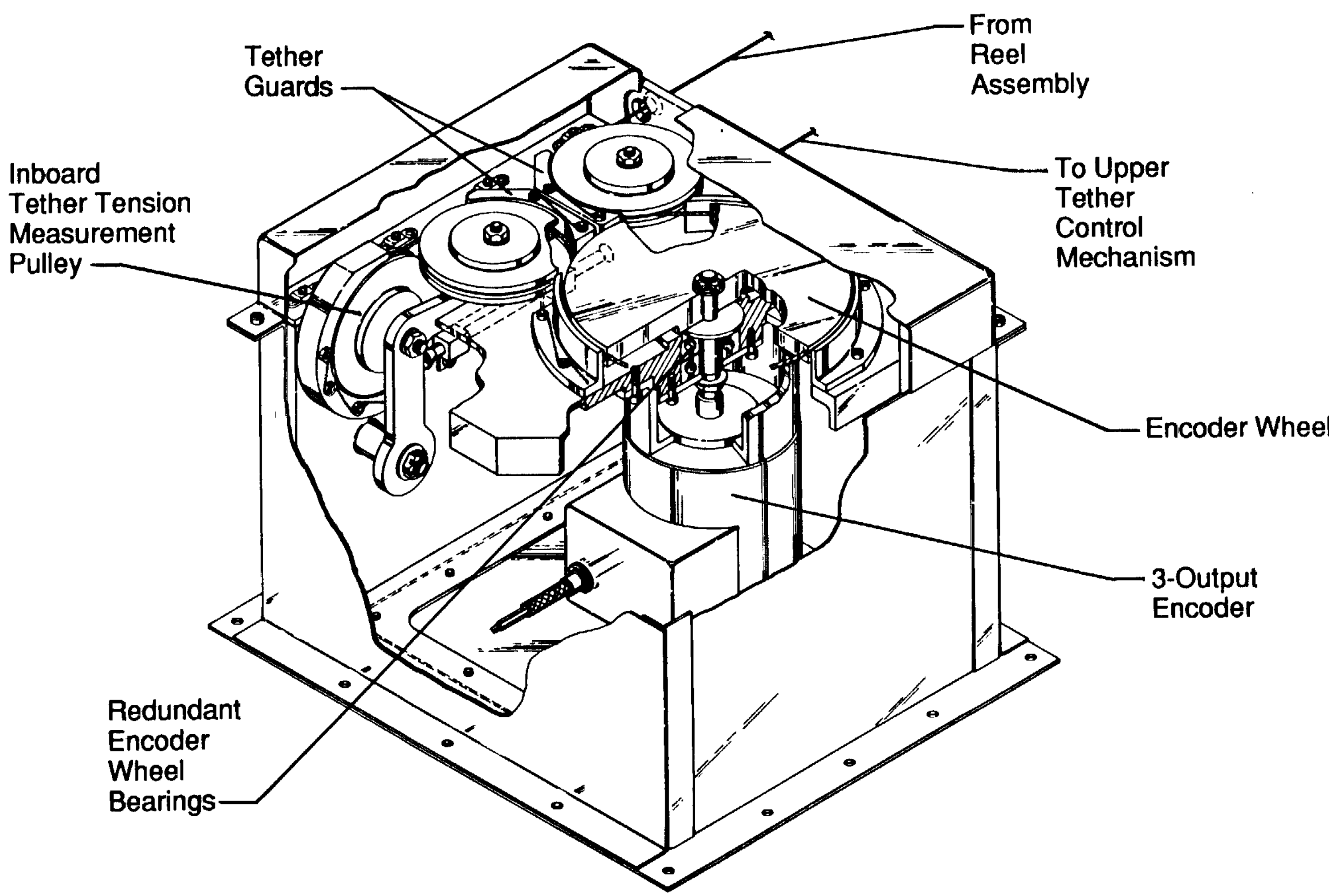
The Lower Tether Control Mechanism (LTCM) Consists of 3 Elements, the Encoder, the Inboard Tensiometer, and Various Tether Guides and Pulleys.

The LTCM is Mounted on the Aft End of the Satellite Support Structure.

In the Retrieval Direction, the Tether Enters the LTCM Through a Guide Tube and Wraps 180° Around the Tensiometer Pulley. The Tether Then Engages the Encoder Pulley with the Help of 2 Guide Pulleys. It Then Departs the LTCM Underneath the Reel Support Structure to the Level Wind Mechanism.

Tether Control Mechanisms

Lower Tether Control Mechanism



Tether Control Mechanisms

Encoder

The Encoder is an Incremental Optical Shaft Encoder That Provides Tether Length and Length Rate Data.

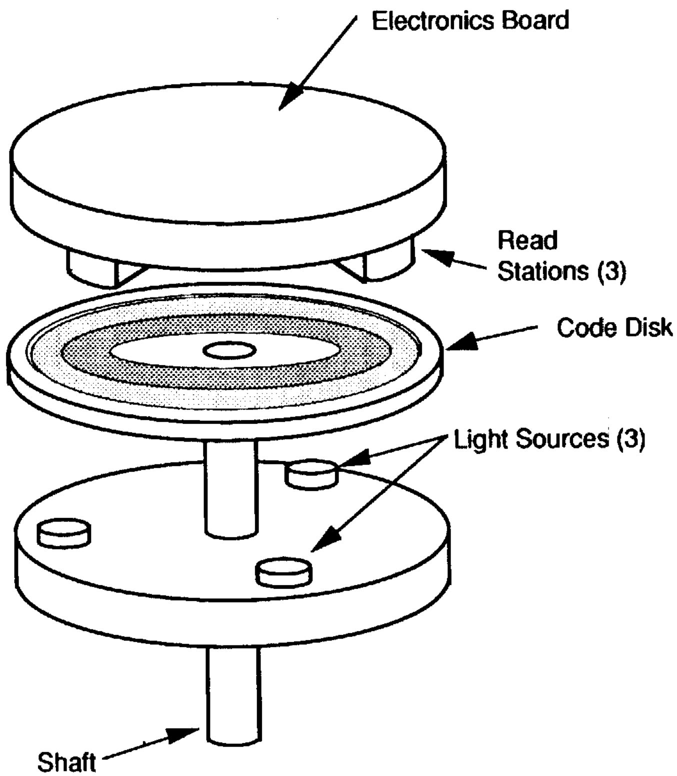
The Tether Takes a 250° Wrap Around the Encoder Pulley Which is Connected to the Encoder by a Shaft. The Shaft Rotates a Slotted, 100 Division-per Revolution Code Disk. 3 Identical read Stations are Located on the Encoder Assembly, 2 Sending Signals to the MCA and 1 Sending Signals to the DACA.

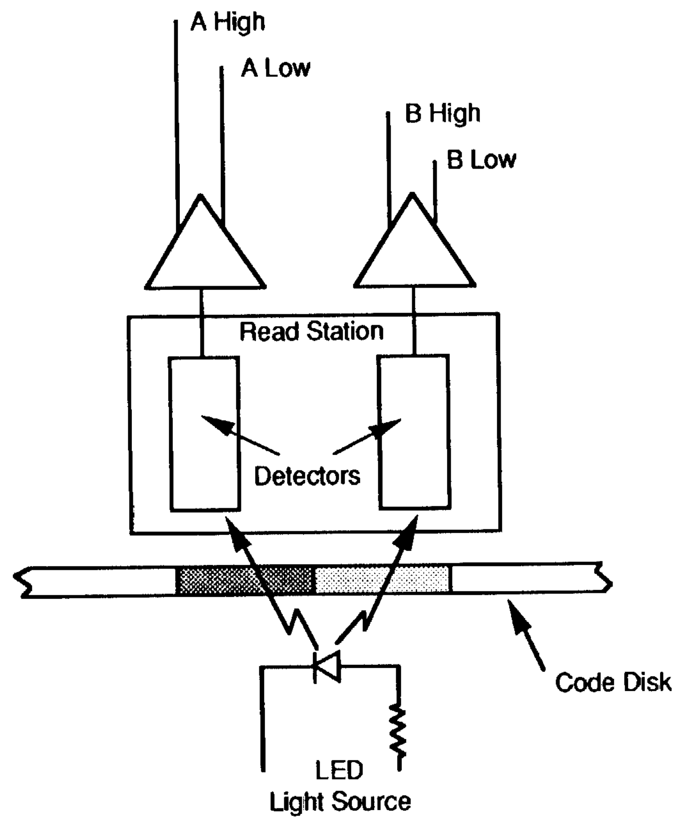
Each Read Station Contains a LED and 2 Detectors. The Slots in the Code Disk are Offset by 90° in Phase So That Direction Can Be Determined. Each Read Station Contains Electronics to Condition and Amplify the Generated Square Wave Signal.

Encoder Resolution is 3.6° of Rotation Which Corresponds to 5.000 mm (0.19 in.) of Tether Movement.

Tether Control Mechanisms

Encoder





Tether Control Mechanisms

Inboard Tensiometer

The Inboard Tensiometer Consists of a Strain Gauge, Bridge Type Load Cell That is Used to Measure the Tether Tension Inboard of the Upper Tether Control Mechanism.

As the Tether Goes Through the LTCM it Wraps Around a Pulley Which is Mounted to a Pivoted Beam. The Beam is Attached to the Load Cell and Structure Through 2 Ball Rod Ends. As the Tension in the Tether Changes, the Resulting Strain in the Load Cell Causes a Resistance Change in the Load Cell Elements. An Excitation Voltage is Applied to the Bridge. The Resistance Change in the Bridge Elements Results in an Output Voltage Proportional to the Strain. This Signal is Conditioned and Digitized by the DACA for Telemetry and the Control Software.

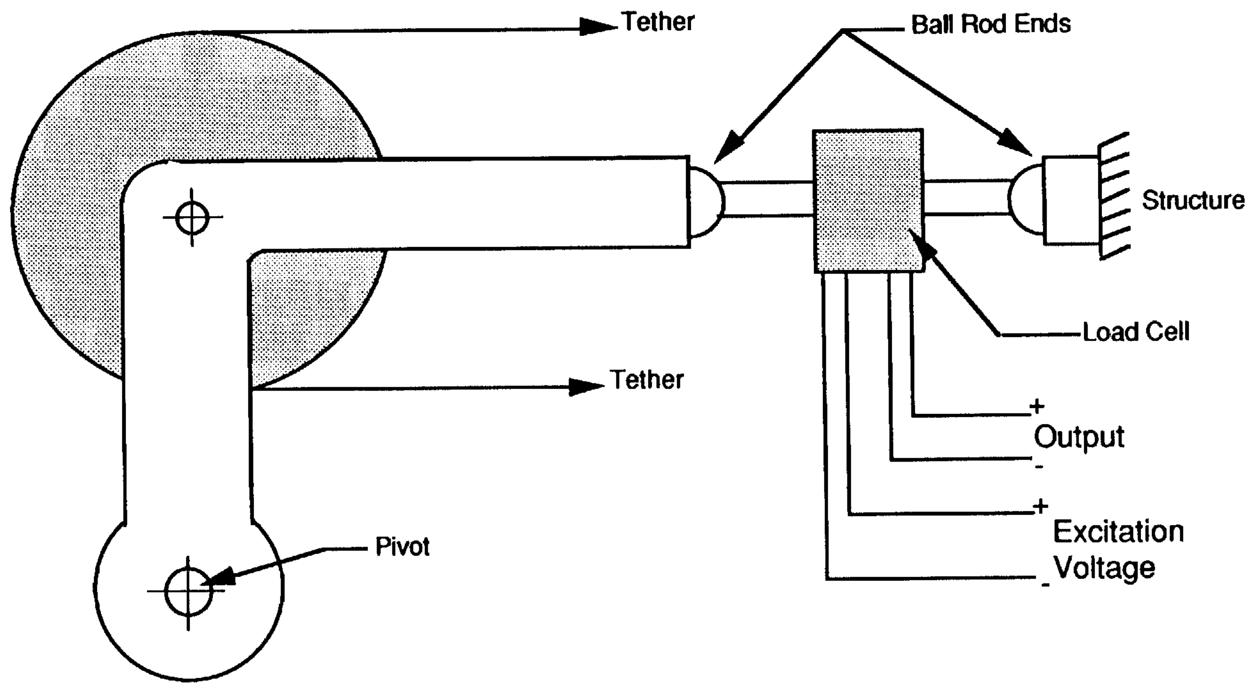
The Tension is Used to Verify System Operation and May Be Used as Feedback to Control Reel Motor Torque with Respect to the Vernier Motor Torque During Deployment.

The Inboard Tensiometer Characteristics are as Follows:

Range	0-110 N
Accuracy	2% of Full Scale Plus 4% of Reading

Tether Control Mechanisms

Inboard Tensiometer



Tether Control Mechanisms

Upper Tether Control Mechanism

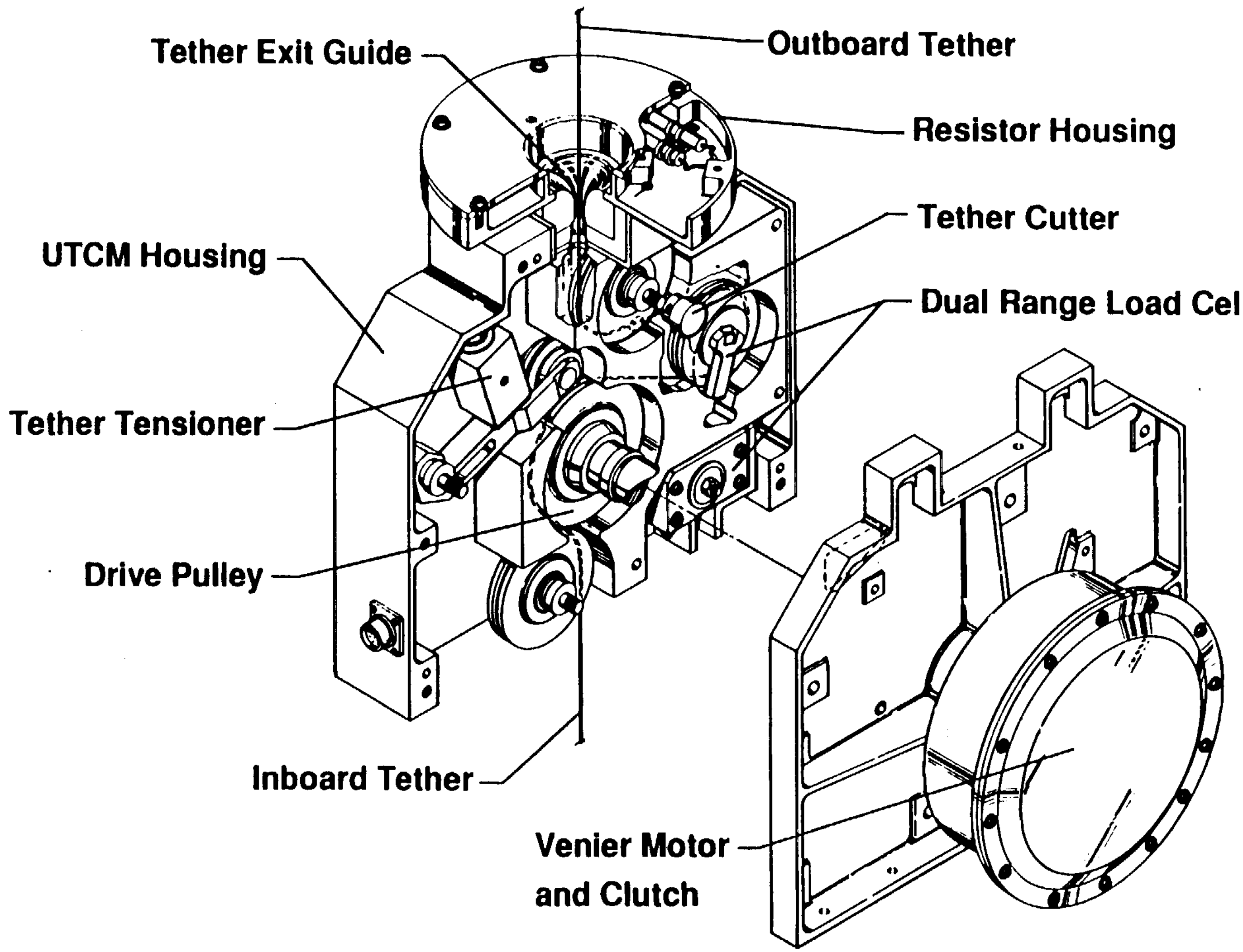
The Upper Tether Control Mechanism (UTCM) Consists of 3 Elements, the Vernier Drive, the Outboard Tensiometer, the Upper Tether Cutter, and Various Tether Guides and Pulleys.

The UTCM is Mounted in the Tip Cannister at the Top of the Satellite Deloyment Boom.

In the Retrieval Direction, the Tether Enters the UTCM Through the Tether Bugle. The Bugle is a Smooth, Conductive, Ceramic Bugle Shaped Guide Which Routes the Tether inot the UTCM and Also Conducts Electrostatic Charge Off the Tether. The Tether then Wraps 90° Around a Pulley and Passes Through the Upper Tether Cutter. It Then Wraps 180° Around the Tensiometer Pulley and Then Engages the Vernier Drive Pulley. The Tether Then Departs the UTCM Down the Center of the Boom.

Tether Control Mechanisms

Upper Tether Control Mechanism



Tether Control Mechanisms

Vernier Drive

The Vernier Drive Consists of the Vernier Motor and Electromagnetic Clutch.

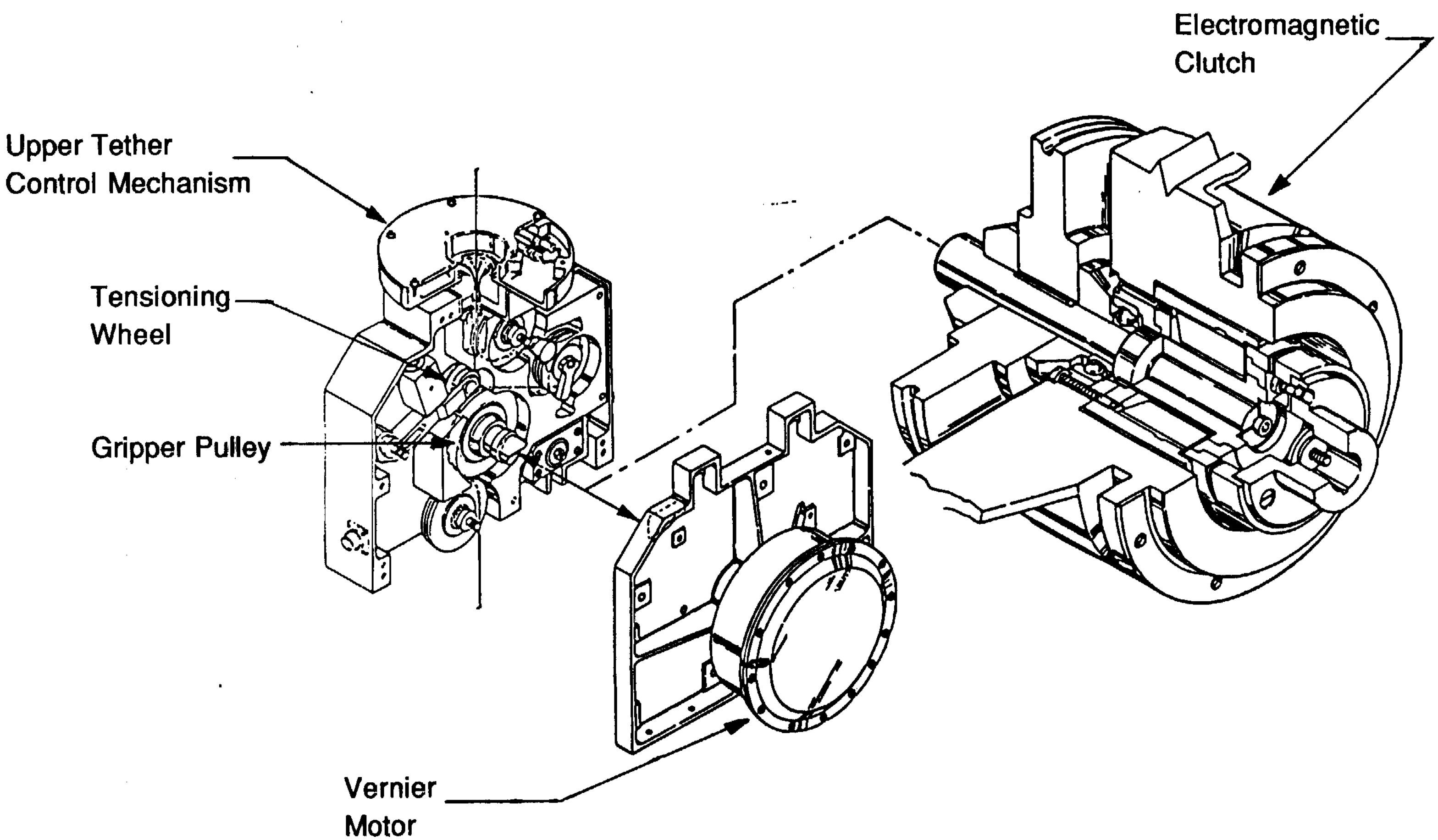
The Vernier Motor is a Brushless DC Motor With a Controller. 3 Position Sensors Within the Motor Provide Rotor Postion Signals to the Controller. The Controller Commutates Power to the Respective Motor Windings Based on the Position Sensor Signals. The Controller Unit Receives Signals From the MCA to Power On, Off, and Rampdown the Motor. The Rampdown Function Linearly Decreases the Motor Voltage Over a 40 Minute Period and Powers Off the Motor. The Clutch is Disengaged to Allow Free Rotation. The Motor's Nominal Torque is 1.41-1.70 N-m (200-240 in-oz).

The Clutch Engages Electromagnetically When DC Power is Applied. It is an In-Line Coupling, Spring Released Device.

The Clutch Drives a Gripper Pulley Which the Tether Rides On for 90°. To Ensure Traction, a Tensioning Wheel is Used. This Consists of a Spring Forced Plunger with a Wheel on the End That Clamps the Tether Between Itself and the Gripper Pulley.

Tether Control Mechanisms

Vernier Drive



Tether Control Mechanisms

Outboard Tensiometer

The Outboard Tensiometer is a 2 Stage, Strain Gauge Bridge Type Load Cell to Provide a High Resolution Outboard Tension Measurement.

As the Tether Goes Through the UTCM it Wraps Around a Pulley Which is Mounted to a Pivoting Beam. Two Load Cells are mounted on the Beam to Detect Strain in the Beam. The 1st Load Cell is the Fine Tensiometer. It Provides Tension Data in the 0 to 9 N Range by Detecting Strain in the Cantilevered Portion of the Beam. The 2nd Load Cell is the Coarse Tensiometer. It Provides Tension Data for the Full Range of 0 to 60 N by Detecting Strain in the Main Portion of the Beam.

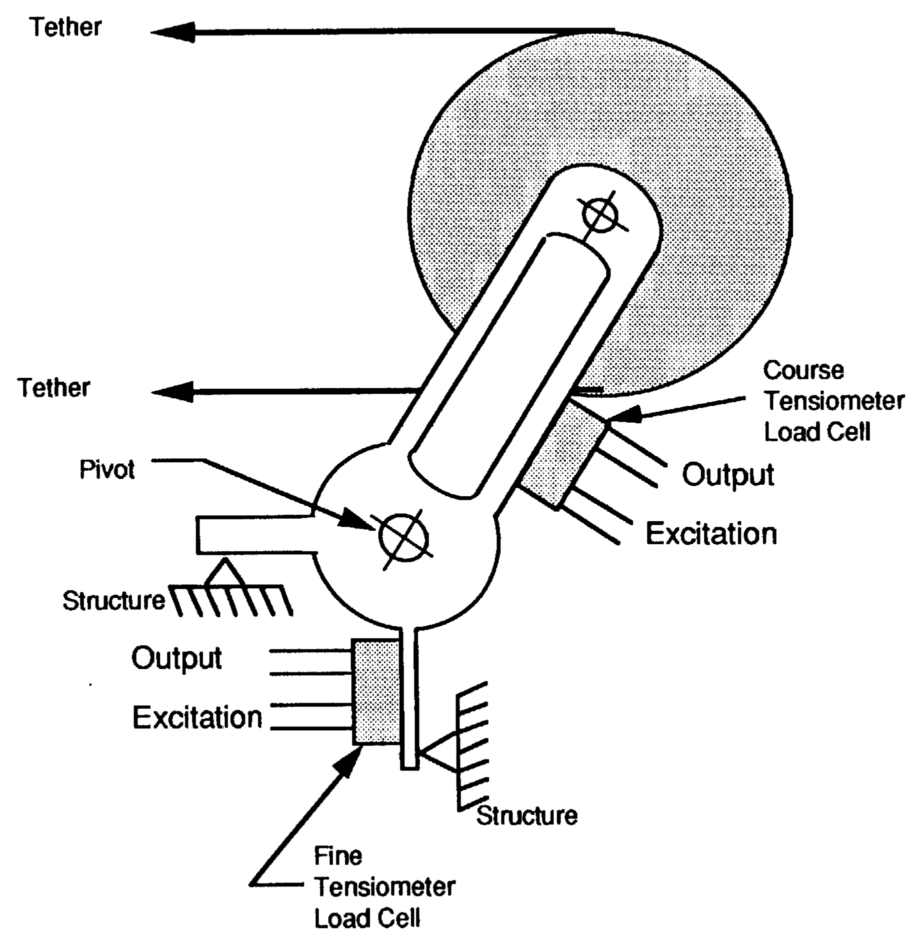
The Tension Data is Used to Verify System Operation During Deployed Operations.

The Outboard Tensiometer Characteristics are as Follows:

Range	
Fine	0-9 N
Coarse	0-60 N
Accuracy	2% of Full Scale Plus 4% of Reading

Tether Control Mechanisms

Outboard Tensiometer



Tether Control Mechanisms

Tether Cutters

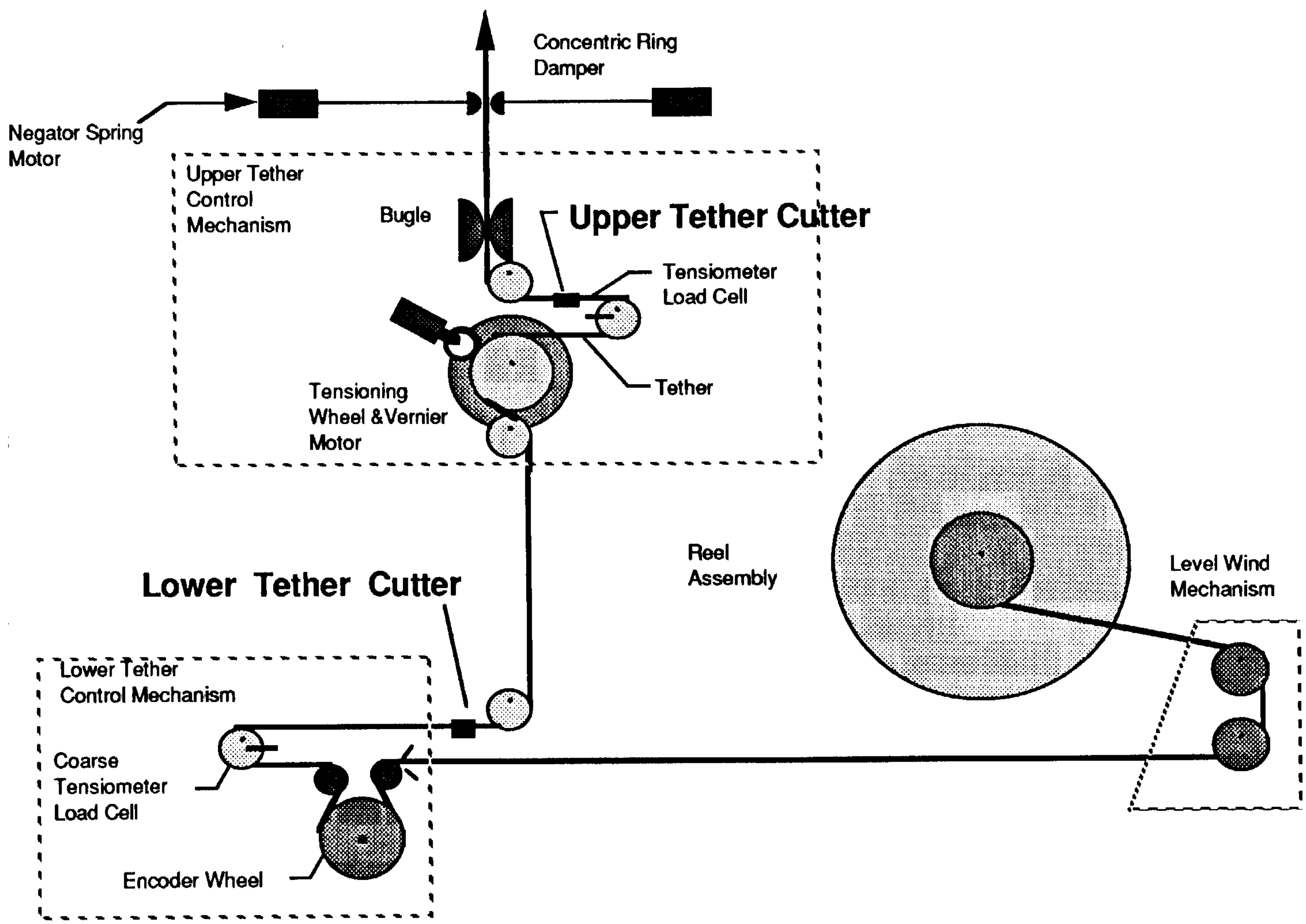
The Deployer Contains 2 Tether Cutters, Which are Refered to as the Upper and Lower Tether Cutters. The Upper Tether Cutter is Mounted in the UTCM. The Lower Tether Cutter is Mounted Between the LTCM and the Idler Pulley at the Base of the Boom.

The Tether Cutters Utilize Gas Pressure from a NASA Standard Initiator (NSI) to Drive a Piston With an Attached Cutting Blade Through the Tether and Into an Anvil. The Anvil Acts to Retain and Support the Tether During Cutting. The Cutting Blade is Retained in the Tether Cutter Body by the Anvil After Cutting the Tether. After Cutting, Both Sections of the Tether are Free for Withdrawal from the Cutter.

Each Tether Cutter Can Be Actuated from Either of 2 Redundant NSIs.

Tether Control Mechanisms

Tether Cutter Locations



Tether Control Mechanisms

Concentric Ring Damper

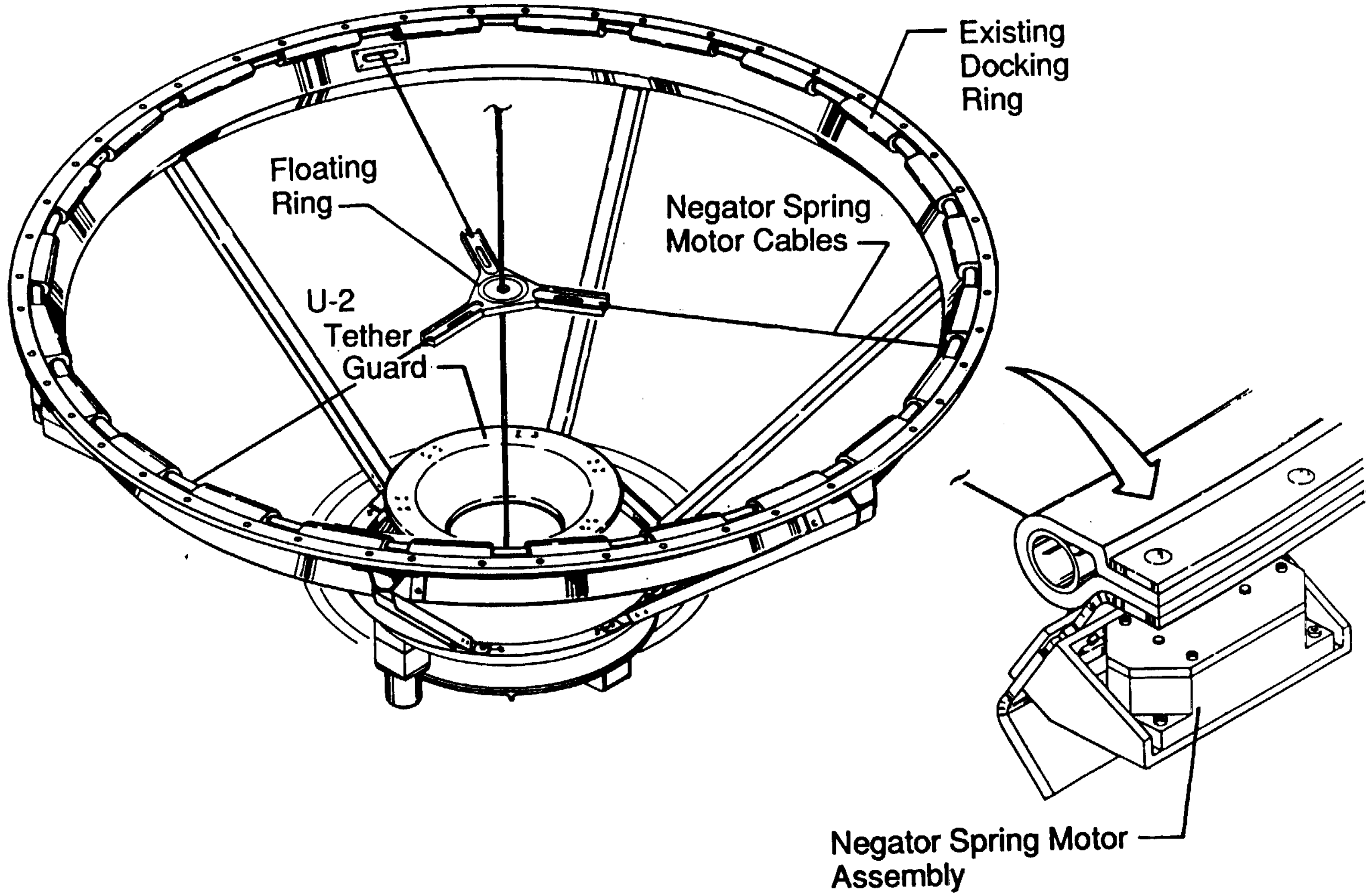
The Concentric Ring Damper is a Mechanical Device That Provides Damping of Tether Skiprope Motion to Maintain Control of the Satellite and Tether System. The Concentric Ring Damper is Located on the Docking Ring.

The Concentric Ring Damper Consists of 3 Negator Spring Motors, 3 Cables, and the Concentric ring. The Tether Passes through the Ring Which is Suspended by the Cables in the Plane of the Docking Ring. As the Tether Moves in a Skiprope Motion the Ring will Move with It. This Motion is Reacted by the Cables Which Are Connected to the Negator Spring Motors. The Negator Spring Motors Have a Hysteresis Which Results in Energy Being Absorbed as the Cable is Cycled In and Out.

The Docking Ring/UTCM also Have a Tether Guard Over the U2 Umbilical Mechanism to Prevent the Tether from Becoming Entangled with the U2.

Tether Control Mechanisms

Concentric Ring Damper



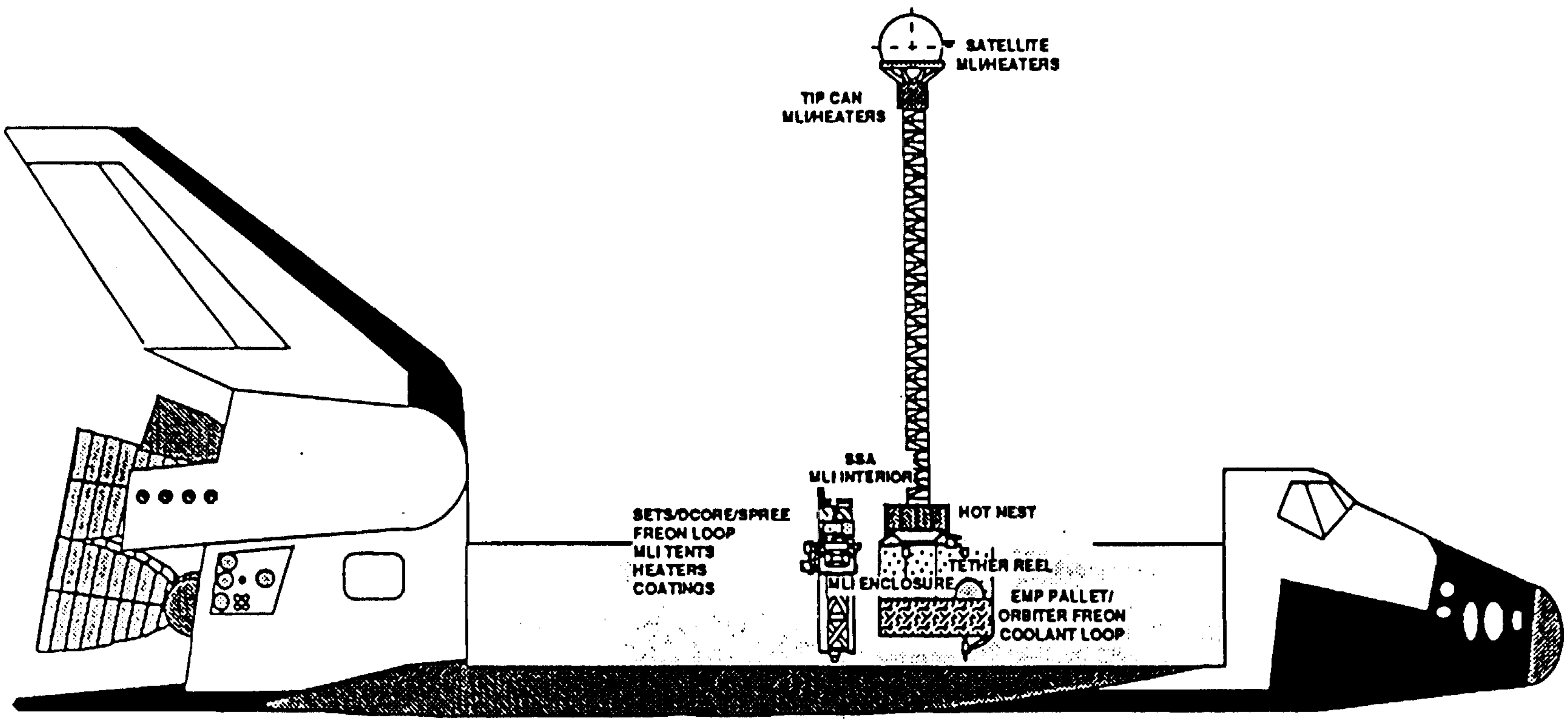
Thermal Control System

TSS-1 Thermal Design Overview

The TSS-1 Components are Shown Mounted in the Orbiter. The Deployer Thermal Design Uses Multilayer Insulation (MLI), Coatings, Heaters, and the Pallet Freon™ Loop. The Tip Canister (TPC) that is Mounted on the End of the Boom Utilizes a Cold Biased Thermal Design. Thermostatic Heaters Maintain Temperatures for Cold Conditions. The MPESS Mounted Science Equipment is Thermally Controlled Using the Pallet Freon™ Loop in Conjunction with MLI. The Satellite Thermal Design is Similar to the TPC Using MLI and Thermostatic Heaters.

Thermal Control System

TSS-1 Thermal Control System Components



Thermal Control System

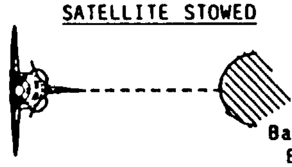
TSS Thermal Analysis Conditions

TSS-1 Requires that the Payload Withstand the Thermal Attitudes Shown. These Conditions have Been Selected to Bound the TSS-1 Thermal Performance. TSS-1 Design Temperature Requirements Vary for Each Mechanism, but a Typical Range is -50 C to +50 C. TSS-1 Design Thermal Environments Encompass Full Solar Exposure to Zero Heat Flux Cold Soak Attitudes.

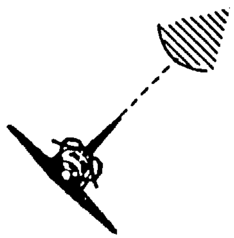
Thermal Control System

TSS Thermal Analysis Conditions

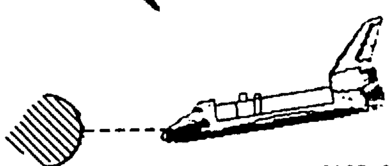
SATELLITE STOWED



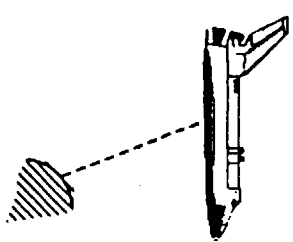
CASE 1
Bay to Earth
Beta = 0°
Cold
Continuous
Case 1A, HN on
(HN = Hot nest)



CASE 2
Bay to Earth
Beta = 52°
Hot
Continuous
Case 2A, HN on



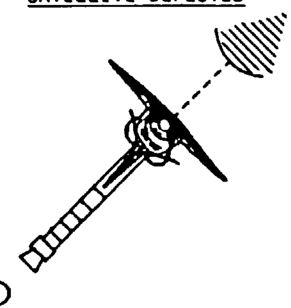
CASE 3
Deep space viewing (pitch down tail to Sun)
Beta = 0°
Cold, 1.5-hr transient/
8-hr recovery
Case 3A, HN on



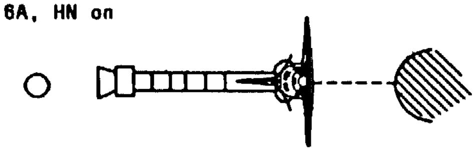
CASE 4
Solar inertial
Beta = 52°
Hot, 0.5 hr
Transient/
6-hr recovery
Case 4A, HN on

SATELLITE DEPLOYED

CASE 5
Bay to space
Beta = 52°, cold
Pitch up 30°
Continuous
Case 5A, HN on



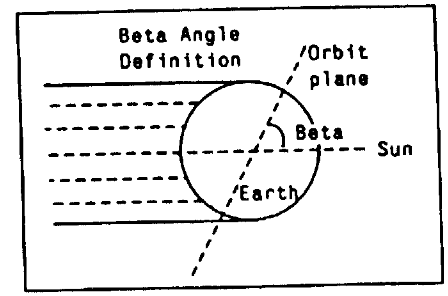
CASE 6
Bay to space
Beta = 0°, hot
Pitch up 30°
Continuous
Case 6A, HN on



Hot Nest Operational Cases
(Hot nest on/satellite stowed)

CASE 7
Bay to Earth, beta = 0°
Cold

CASE 8
Bay to Earth, beta = 52°
Hot



Prelaunch/Ascent/Entry	
Case	Description
11	Prelaunch/ascent cold
12	Prelauch/ascent hot
13	Entry/postlanding cold, no purge
14	Entry/postlanding hot, no purge
14a	Entry/postlanding hot with purge

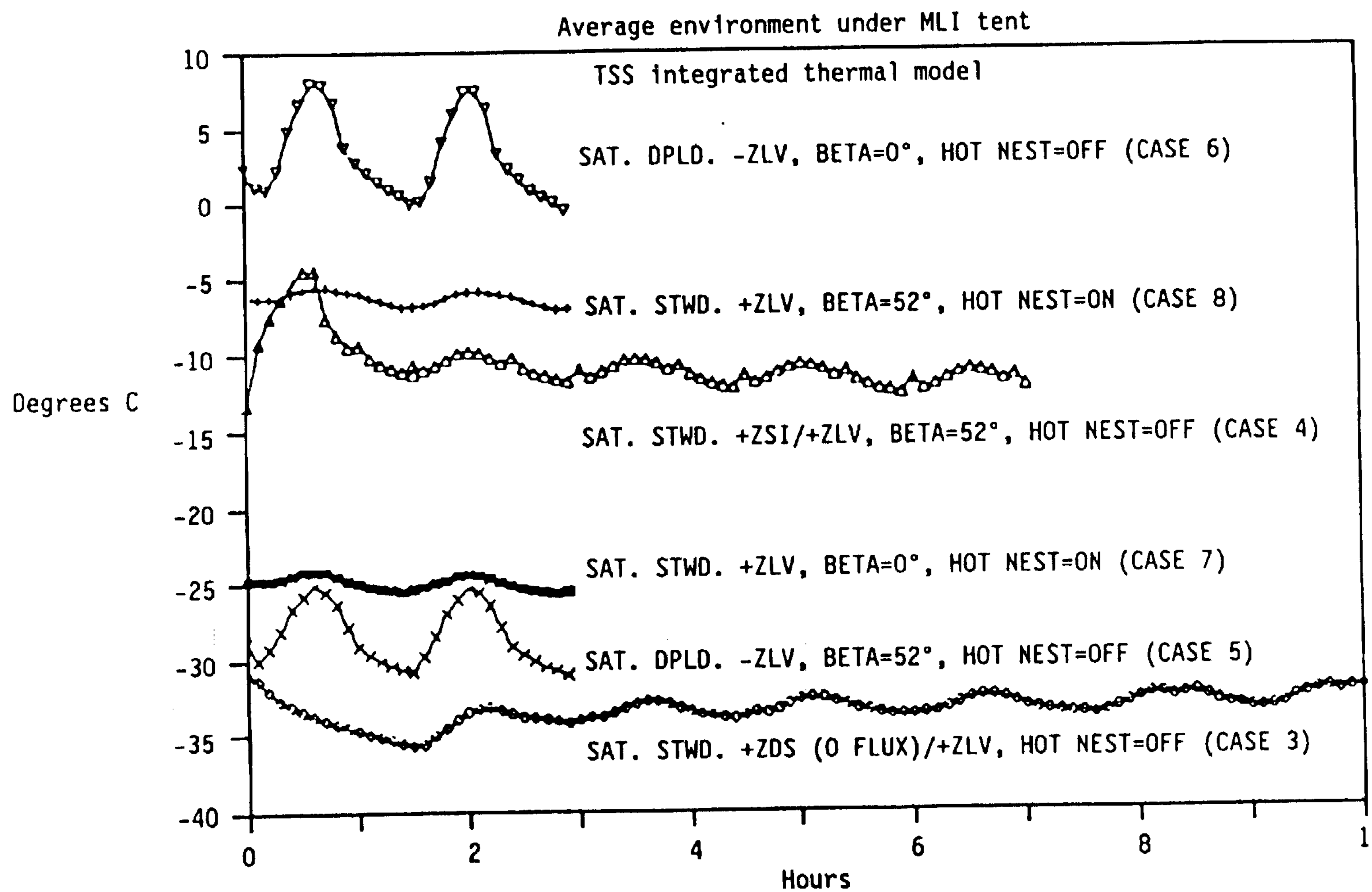
Thermal Control System

On-Orbit Deployer Temperature Predictions

Representative Predicted Temperatures for the Radiative Thermal Environment Beneath the Deployer Thermal enclosure are Shown for Selected On-Orbit Conditions. Temperatures Vary Between -35 C and +10 C. The Temperature Variations Shown are a Result of Orbital Heat Flux Variations.

Thermal Control System

On-Orbit Deployer Temperature Predictions



Thermal Control System

TSS-1 Deployer Thermal Design

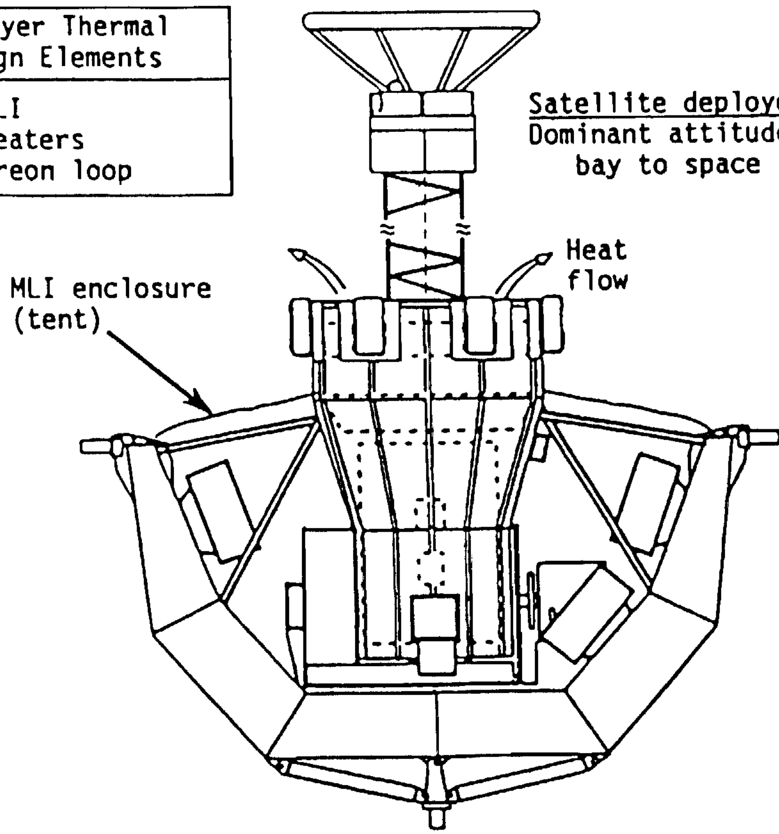
Active and Passive Thermal Design Approaches are Used for Both the Deployer, EMP, and MPESS Mounted Science Equipment. The TSS Payload Uses the EMP Freon™ Loop, Various Heater Systems, and Multilayer Insulation (MLI) to Maintain Payload Temperatures Within Limits.

The TSS Thermal Design Uses a Unique MLI Approach. Components that are Mounted on the Pallet are Isolated form the Space Thermal Environment by Using a Large MLI Enclosure that Covers all TSS Pallet Mounted Components. This Approach Simplifies the TSS Thermal Design, Thermal Analysis, and MLI Fabrication. This is the First Application of a Pallet MLI Enclosure to a Spacelab Pallet.

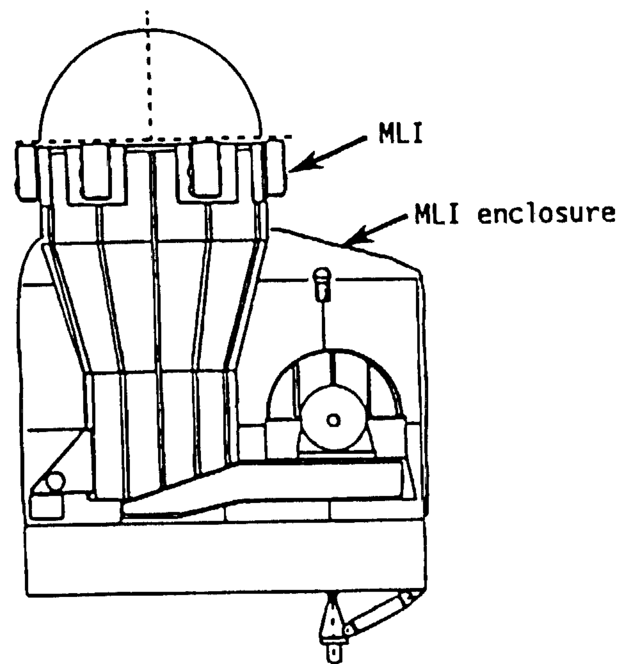
Thermal Control System

TSS-1 Deployer Thermal Design

Deployer Thermal Design Elements
• MLI • Heaters • Freon loop



Looking aft



Side view

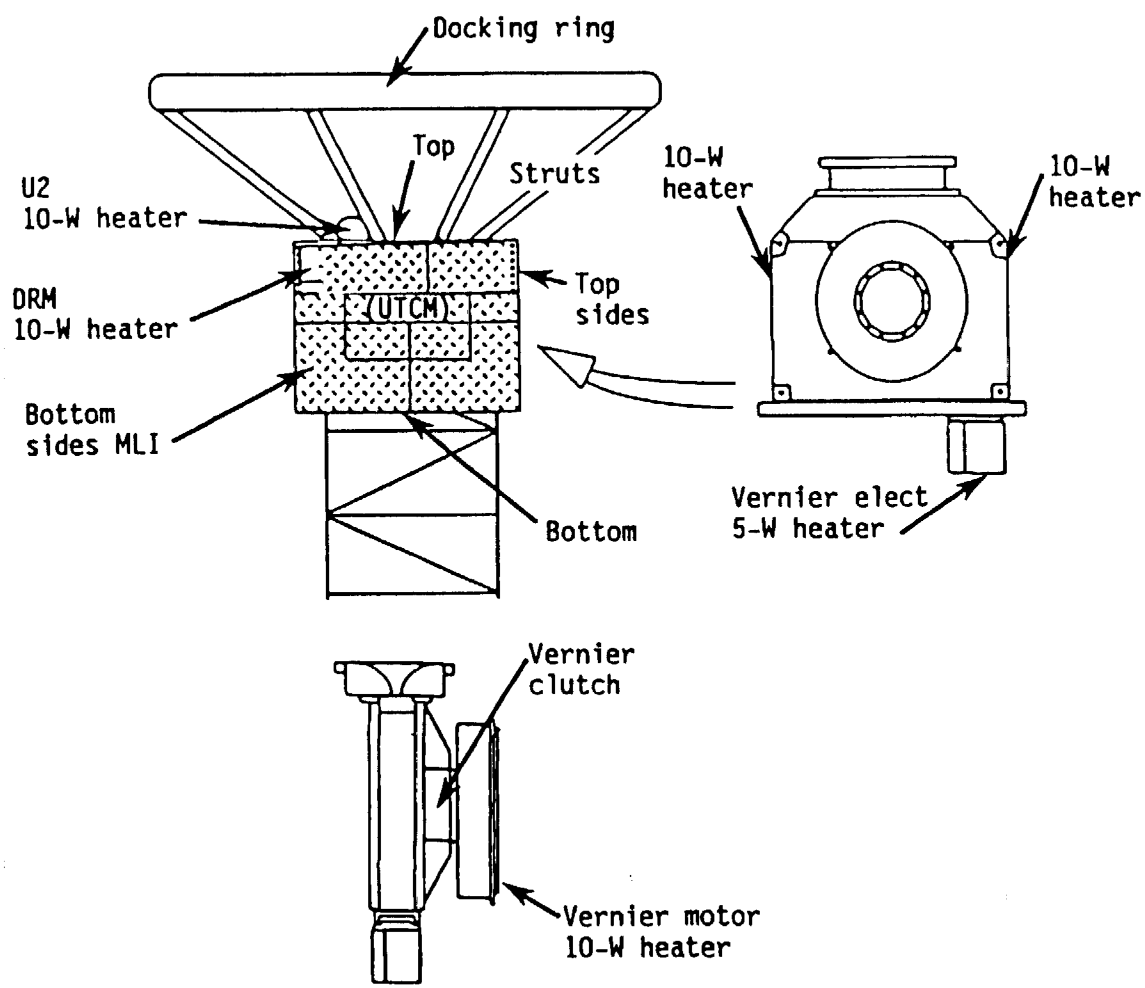
Thermal Control System

Tip Canister Thermal Design

The Tip Canister Thermal Design Uses a MLI on External Surfaces and Electrical Heaters on Five Key Internal Components. The External TPC Structure is Painted White and Acts as a Thermal Radiator. MLI Covers Most of the External Surface Except for the Top, that is a Combination of White Paint and Aluminum Tape. The External Layer of the MLI is Indium Tin Oxide (ITO) Coated Kapton Because of the Electrical Conductivity Requirement.

Thermal Control System

Tip Canister Thermal Design



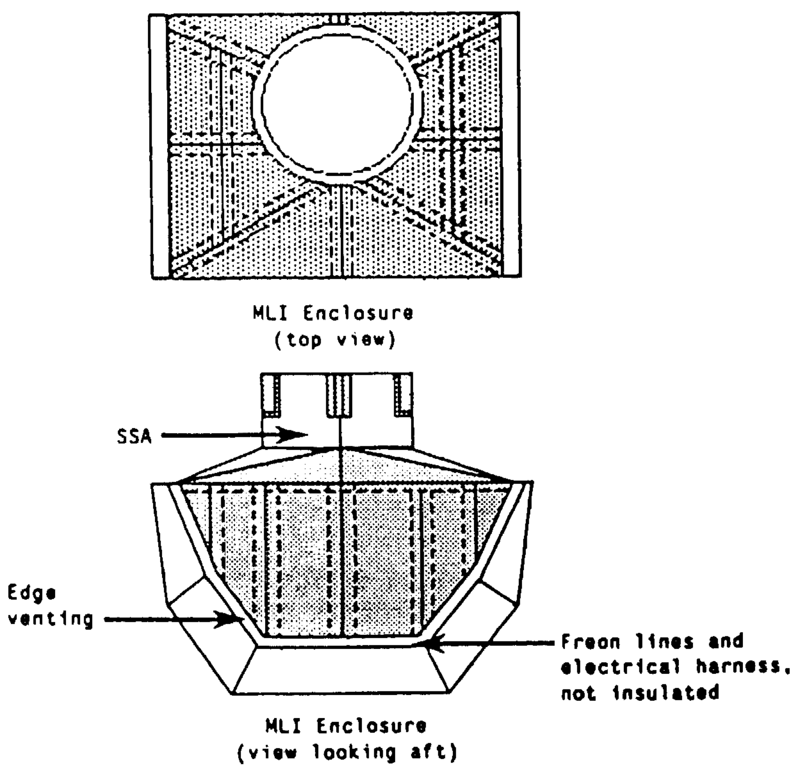
Thermal Control System

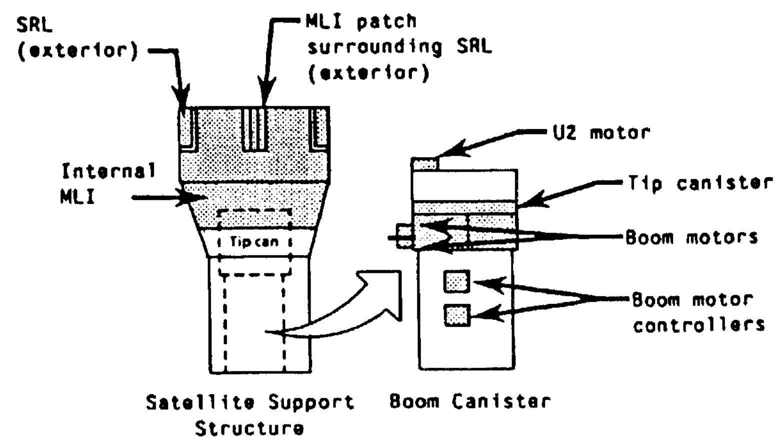
Deployer Multilayer Insulation Locations

The Following Chart Shows the Detailed Configurations of the Deployer MLI. Aluminized Layers of Kapton or Mylar are Formed into Blankets that Consist of Multiple Layers. The Low Emissivity of the Layers Inhibit Radiative Heat Flow, Isolating the Components from the Space Thermal Environment. The MLI Blankets are Grounded and Vented, Both Individually and at the Enclosure Level. The MLI Enclosure is Supported by a Kevlar Strap System. All MLI is Held in Place by at Least Two Methods, Primarily Velcro and String Tie Downs.

Thermal Control System

Deployer Multilayer Insulation Locations





Shading indicates multilayer insulation

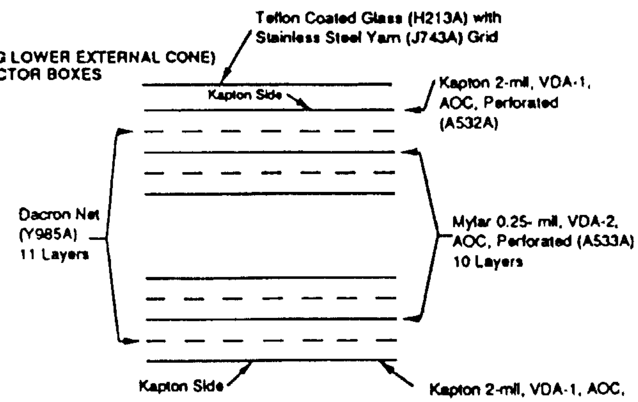
Thermal Control System

Deployer MLI Layup

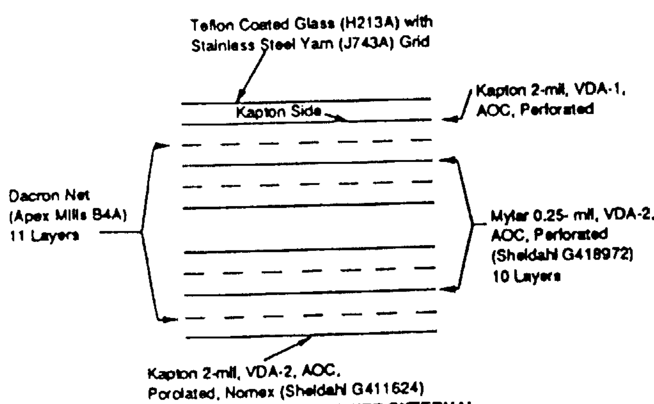
MLI is the Prime Thermal Subsystem Material. The Aluminized Layers of Kapton and Mylar are Typically Less Than Two Mills Thick. Three Major Blanket Configurations are Used. Electrical; Grounding and Venting are Prime Design Considerations

Deployer MLI Layup

BLANKETS:
U2
BOOM
SSA (EXCLUDING LOWER EXTERNAL CONE)
TIP CAN CONNECTOR BOXES

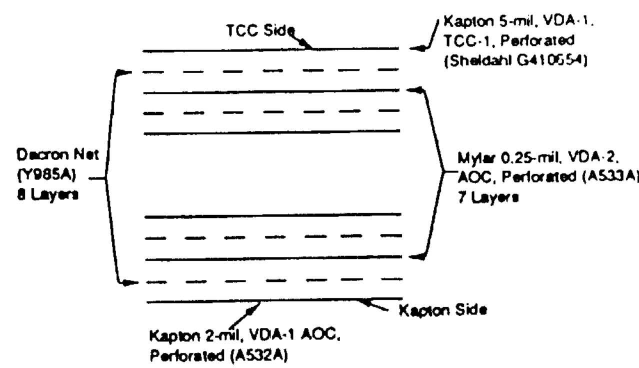


CONFIG-1



TENT, SRL, AND SSA LOWER EXTERNAL CONE BLANKETS

CONFIG-2



TIP CANISTER BLANKETS EXCLUDING CONNECTOR BOXES

CONFIG-3

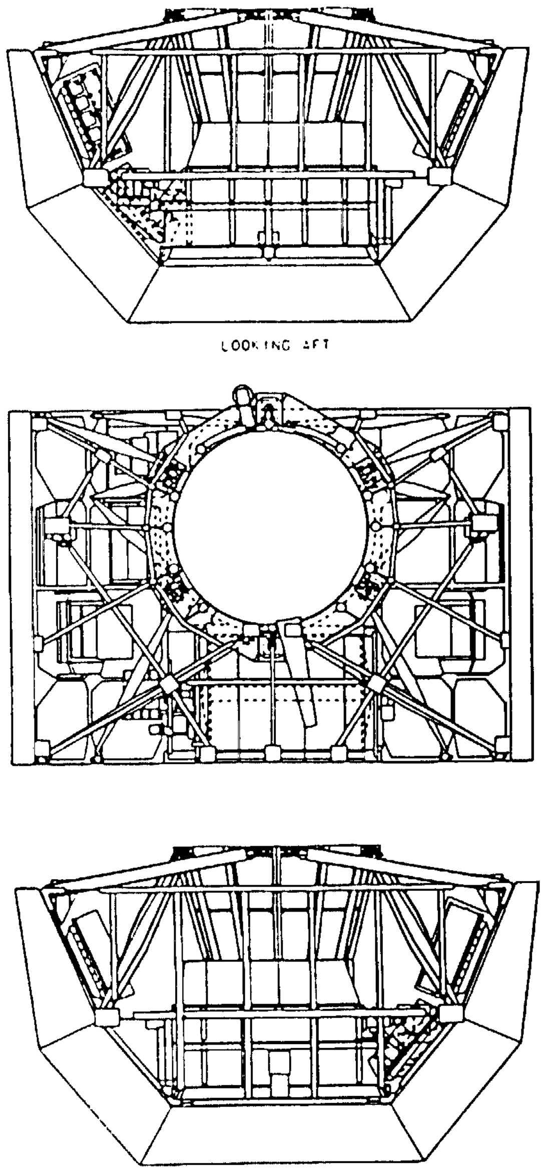
Thermal Control System

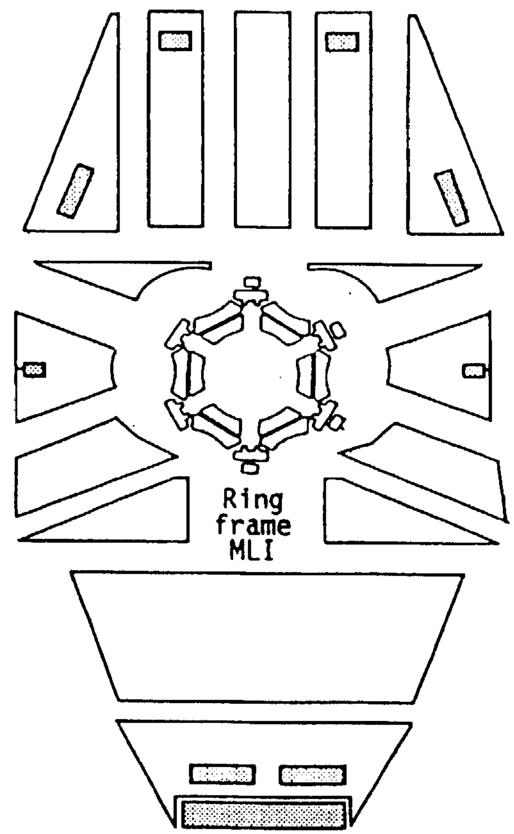
Deployer Thermal Enclosure Configuration

TSS Uses Approximately 56 Square Meters of MLI Blankets. Kevlar Straps Support the Thermal Enclosure as Shown. The Strap System Forms a Loose System to Avoid Dynamic Coupling Between the Enclosure and the Pallet. The Chart Also Illustrates the Enclosure MLI Blanket Components. Venting Areas are Shaded. It has Been a Significant Design Simplification that the Blankets are Flat, and Therefore Easy to Fabricate. Another Advantage of the Thermal Enclosure Blanket Design is that Pallet Build-up Could Proceed Independently.

Thermal Control System

Deployer Thermal Enclosure Configuration





Enclosure/SSA ring frame MLI layout

Thermal Control System

Deployer Heater Descriptions

The Deployer Active Thermal Approach Uses Thermostatically Controlled Kapton Strip Heaters on Critical Mechanisms. Each Heater is Controlled with Two Mechanical Thermostats that are Wired in Series to Prevent a Failed "ON" Condition. Heater Power is Controlled from PCB Relay K2. The Hot Nest Heater is Controlled from PCB Relay K5. The Heaters are not Redundant Except for the LTCM. The Tip Canister Provides a Measure of Heater Redundancy Because of the Five Separate Heated Components. The Chart Details Heater Parameters.

Deployer Heater Descriptions

Ref. no.	Description	Qty	Shape	Power at 22 V (W)	Total power (W)	T'stat qty	T'stat ON T (°C)	T'stat OFF T (°C)	Thermistor qty	Ref. ID
1	LTCMA	1	CYL	10	.10	2 (Note 1)	-36.7	-25.6	1	T20
	LTCMB	1	CYL	10	10	2	-17.2	-12.2	1	T21
2	Latch motors	6	CYL	10	60	12	-17.2	12.2	6	T1-T6
3	Latch relay box	6	N/A	6.3	6.3	12	-36.7	-25.6	6	None
4	Rotation motor	1	CYL	8.3 (Note 6)	8.3	2	-17.2	-12.2	1	T7
5	Boom motor A	1	CYL	10	10	2	-8.3	-3.3	1	T8
	Boom motor B	1	CYL	10	10	2	-8.3	-3.3	1	T9
6	Boom mtr. ctrl. A	1	RECT	5	5	2	-8.3	-3.3	1	T10
	Boom mtr. ctrl. B	1	RECT	5	5	2	-8.3	-3.3	1	T11
7	Vernier motor	1	CYL	8.3 (Note 6)	8.3	2	-3.9	1.1	1	T14
8	Vern. mtr. ctrl.	1	RECT	4.1 (Note 6)	4.1	2	-3.9	1.1	1	T15
9	UTCM A	2	CYL	16.5 (Note 6)	16.5	1 (Note 2)	-3.9	1.1	1	T22
	UTCM B	1	RECT	12.4 (Note 6)	12.4	1 (Note 2)	-3.9	1.1		
10	Load bank	4	RECT	N/A	EDY=100	N/A	N/A	N/A	2	T33-T36
		4	RECT	N/A	ATM=2000	N/A	N/A	N/A	2	
11	Brake motor	1	CYL	10	10	2	-17.2	-12.2	1	T26
		1	RECT	5	5					
12	Hot nest	4	CONE	37.5	300	4 (Note 3)	43.3	46.7	(Note 7)	T38
		4	CONE	37.5		4 (Note 3)				T39
13	HVRA A	2	RECT	5	10	2 (Note 4)	-36.7	-25.6	1	T40
	HVRA B	1	RECT	5	5	2 (Note 4)	-36.7	-25.6	1	T41
14	U1	1	CYL	20	20	2	-17.2	-12.2	1	None
15	U2	1	CYL	8.3 (Note 6)	10	2	-17.2	-12.2	1	None
16	Reel lock	1	CYL	20	20	2	-17.2	-12.2	1	T45
	Totals	35			650.9	42				

Notes:
1. The heater assemblies are configured with two thermostats in series for each heater, except as noted.
2. The three heaters are ganged in parallel and operate off of a common series arrangement of two thermostats.
3. The eight heaters from ref. 11 are wired in parallel sets of two. Each parallel set of two is controlled by a pair of series-wired thermostats.
4. Two of the three HVRA heaters are wired in parallel with a pair of series-wired thermostats in control. The third heater has a separate pair of series-wired control thermostats.
5. Heater power provided by two internal resistors per relay box.
6. Tip canister minimum voltage (20 volts) derived from post-TBT measurements.
7. Thermostat discrete open at 67.8° C.

Thermal Control System

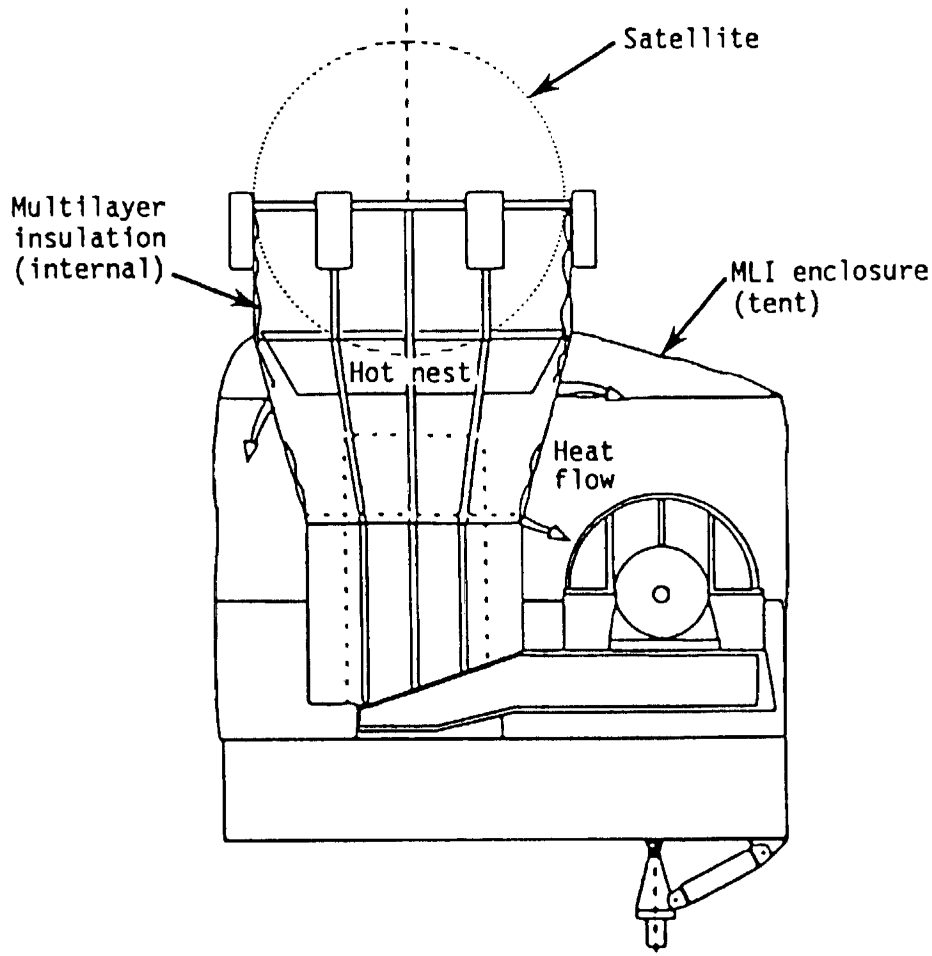
Deployer Hot-Nest Heater Configuration

The Hot-Nest is the Largest and Most Powerful Heater Used in the Deployer Thermal Design. The Hot-Nest Heaters are Mounted on a Conical Aluminum Structure that is in the SSA Below the Satellite. The Hot-Nest is Used to Maintain the Satellite Gyro Above its Lower Storage Temperature During the Post Retrieval Mission Phases.

The Hot-Nest Heater Can Also Warm the Deployer by Acting as a Central Source of Heat Within the Pallet Thermal Enclosure.

Thermal Control System

Deployer Hot-Nest Heater Configuration



Thermal Control System

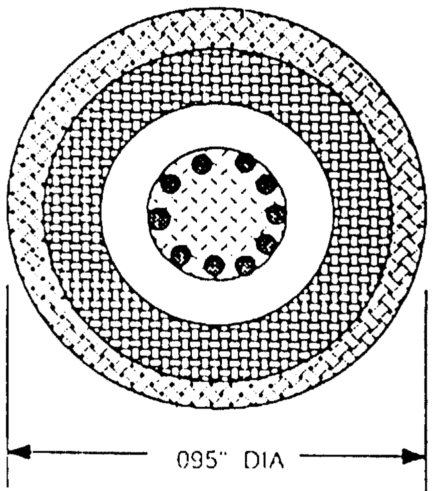
Tether Heating Considerations

The TSS-1 Tether is Electrically Conductive and of Composite Design as shown. During the TSS-1 Mission Electrical Current Flows Through the Tether. The Stowed Tether (Tether on the Reel) is the Limiting Factor for Current Flow Due to Resistive Heating Effects

A Key Thermal Parameter for the Stowed Tether is the Thermal Conductance Between Layers. This Conductance is Difficult to Predict and We Have Completed an Ambient Test to Assess Tether Interlayer Conductance. The Resistive Heating Results in Tether Temperatures Below 125 C (Tether Qualification) for all Mission Phases.

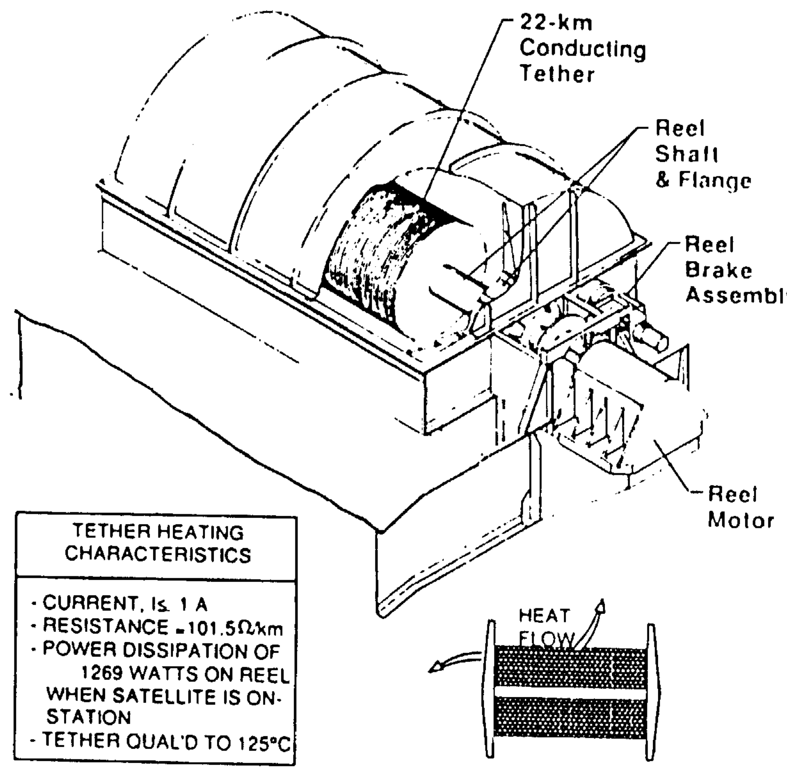
Thermal Control System

Tether Heating Considerations



TSS - 1 CABLE COMPONENTS	
	NOMEX CORE - 2400 DENIER
	10 STRANDS #34 AWG ANNEALED BARE COPPER
	FEP EXTRUDED INSULATION .012" WALL THICKNESS
	KEVLAR STRENGTH MEMBER B29/12 X 1000 DENIER 400 LBS MIN BREAK LOAD
	NOMEX BRAIDED JACKET BNX/8 X 1200 DENIER

Tether Materials



Tether Stowage Reel

Thermal Control System

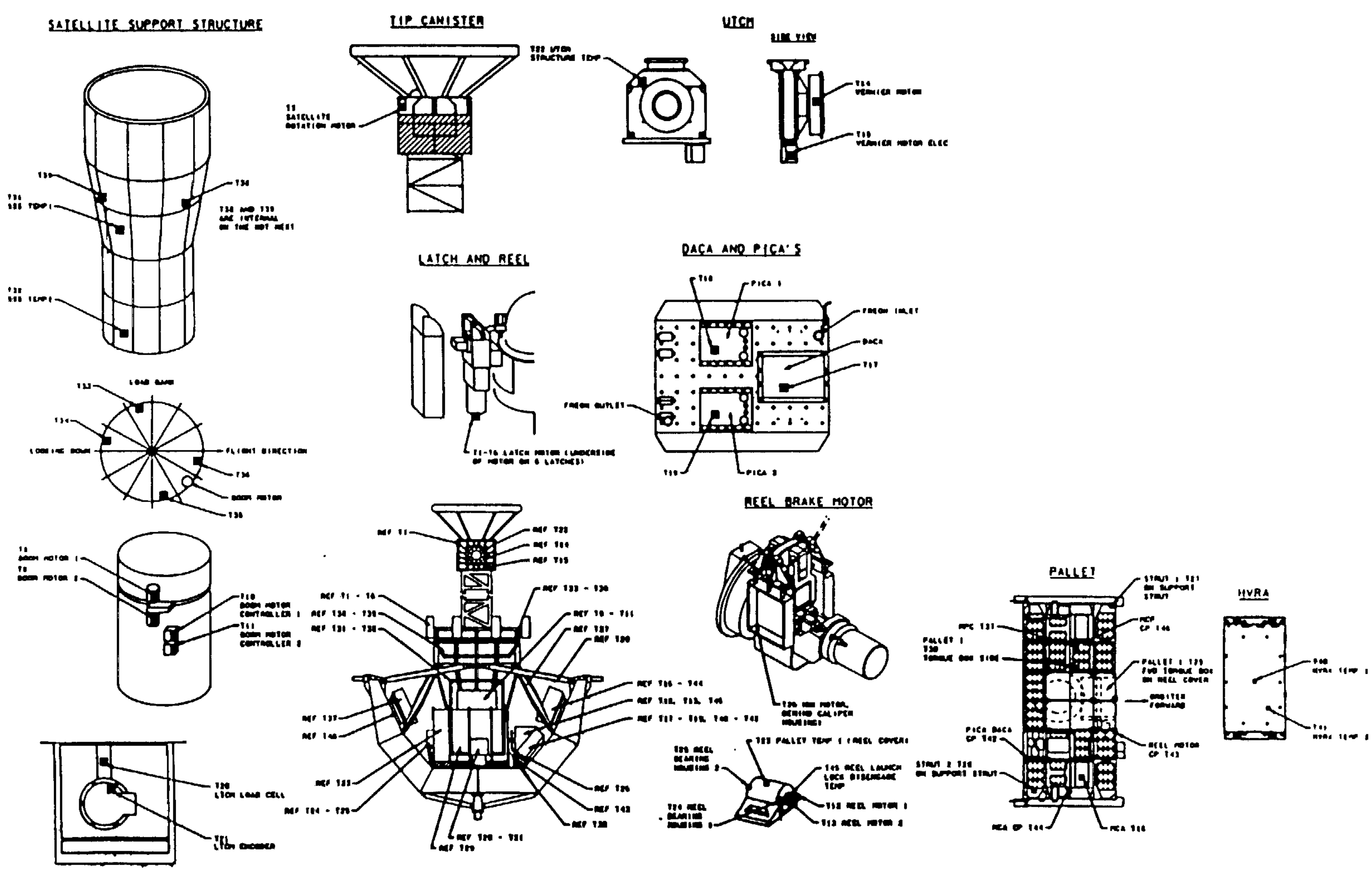
Deployer Flight Thermistor Locations

The Deployer Uses 44 Thermistors to Measure the Temperatures During the TSS-1 Mission. The Locations of These Thermistors are Shown. The Number Used to Identify the Thermistors are MMAG In-house IDs. The Thermistors MSID Numbers and Additional Important Parameters are Shown in the Following Chart.

The Deployer Also has 1 Thermostat to Provide a Discrete Inidication of an Over Temperature Condition on the Hot Nest.

Thermal Control System

Deployer Flight Thermistor Locations



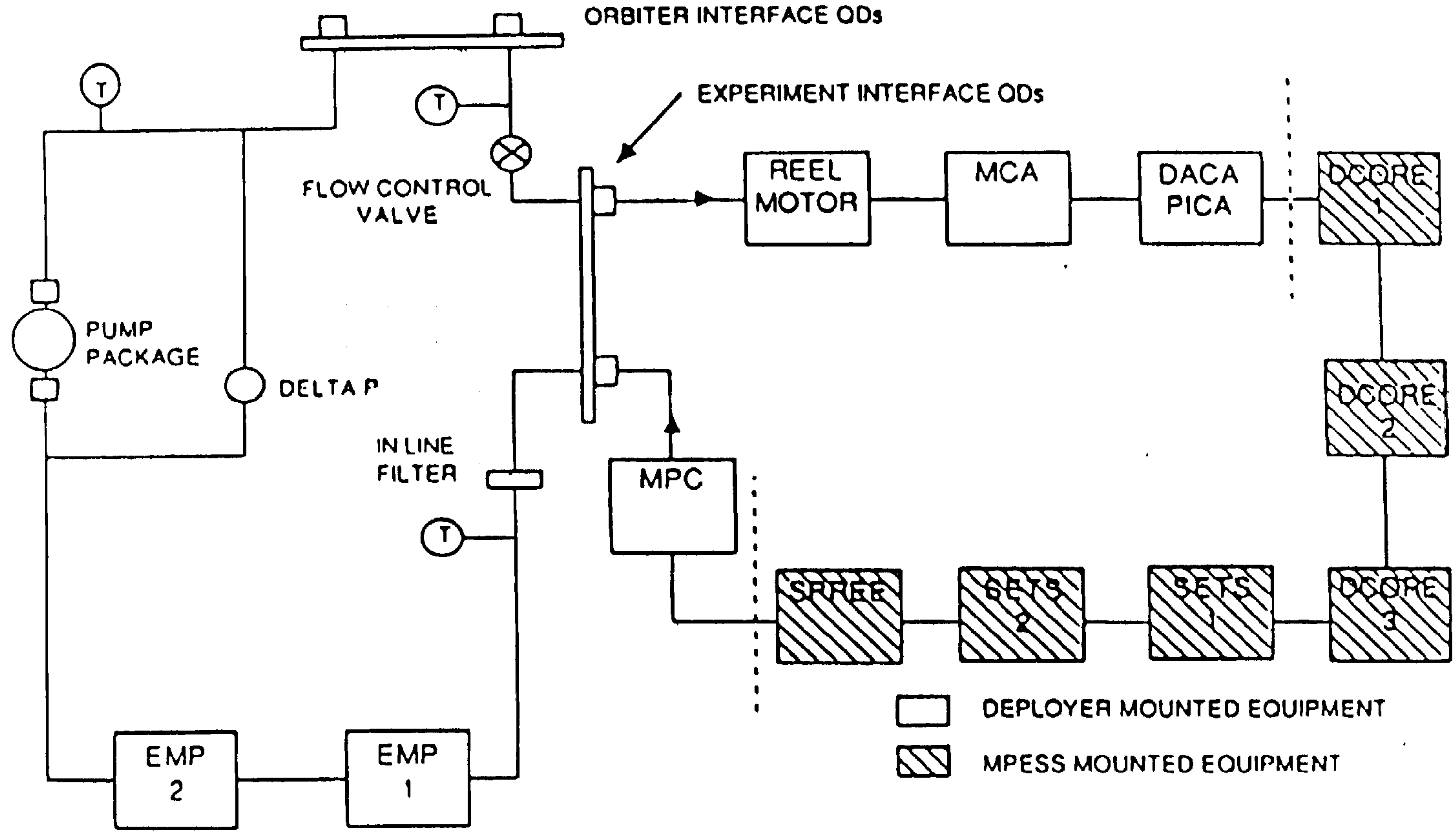
Thermal Control System

Deployer MPESS Freon™ Loop Configuration

The Deployer Freon™ Coolant System is Illustrated. The Loop Configuration Consists of a Single Phase Freon™-114 Design. Coldplates are Used to Transfer Heat from EMP and Deployer Subsystems and MPESS Mounted Science to the Orbiter ATCS Via the Payload Heat Exchanger. Major Components of the ATCS Include a Freon™ Pump Package, Coldplates, a Flow Control Valve that is Set Preflight, Quick Disconnects, Inline Filters, Temperature and Pressure Transducers, and Associated Plumbing. The Pump Package Uses Two Pumps One of Which is Active. The Accumulator Prevents Over-Pressurization and Provides Coolant Makeup for Leakage. The Loop is Designed for 200 psia.

Thermal Control System

Deployer MPESS Freon™ Loop Configuration



Electrical Power Distribution System

Overview

Deployer Power is Provided by the +28 Vdc Orbiter Power Bus. The Enhanced MDM Pallet (EMP) Power Control Box (PCB) Provides Switching Capability for Various Deployer Loads.

Orbiter Auxiliary Power is Provided Directly to the Deployer via EMP Interface Connectors.

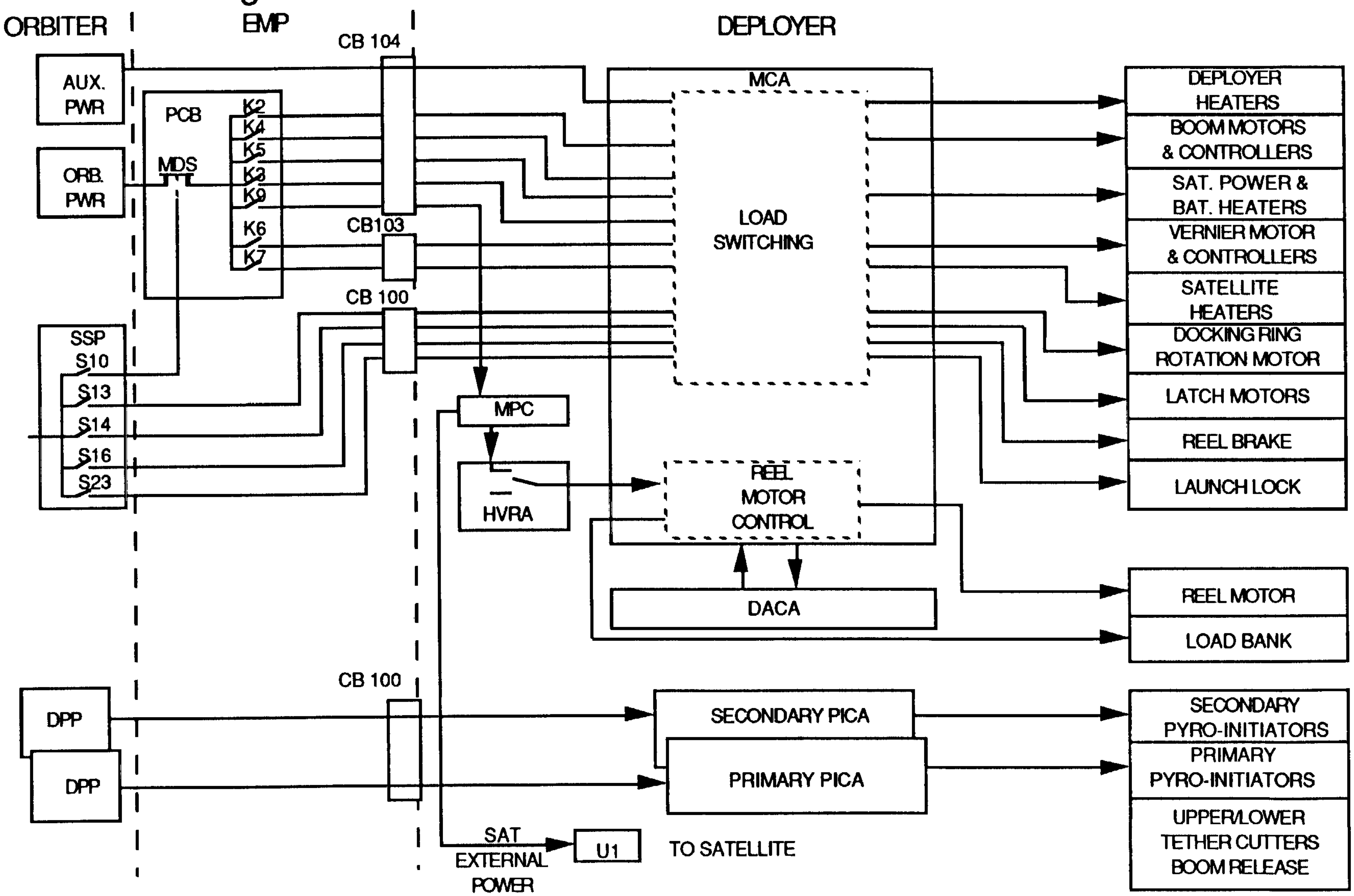
The Electrical Power Distribution System (EPDS) Provides Power Conditioning and Distribution for the Deployer Components During all Mission Phases and Satellite Power During Predeployment Phases.

The EPDS Consists of:

- Interconnecting Cable Harness
- Motor Power Conditioner (MPC)
- Motor Control Assembly (MCA)
- Data Acquisition and Control Assembly (DACA)
- High Voltage Relay Assembly (HVRA)
- Pyrotechnic Initiator Controller Assemblies (PICAs)

Electrical Power Distribution System

EPDS Block Diagram



Electrical Power Distribution System

Interconnecting Cable Harness

The Deployer Cable Harness Routes DC Power, and Discrete and Analog Signals Between Deployer Components and from Interface Connectors on the EMP to the Deployer Components.

The Harness has 2 Major Segments Which are Separated by an Interface Connector Bracket:

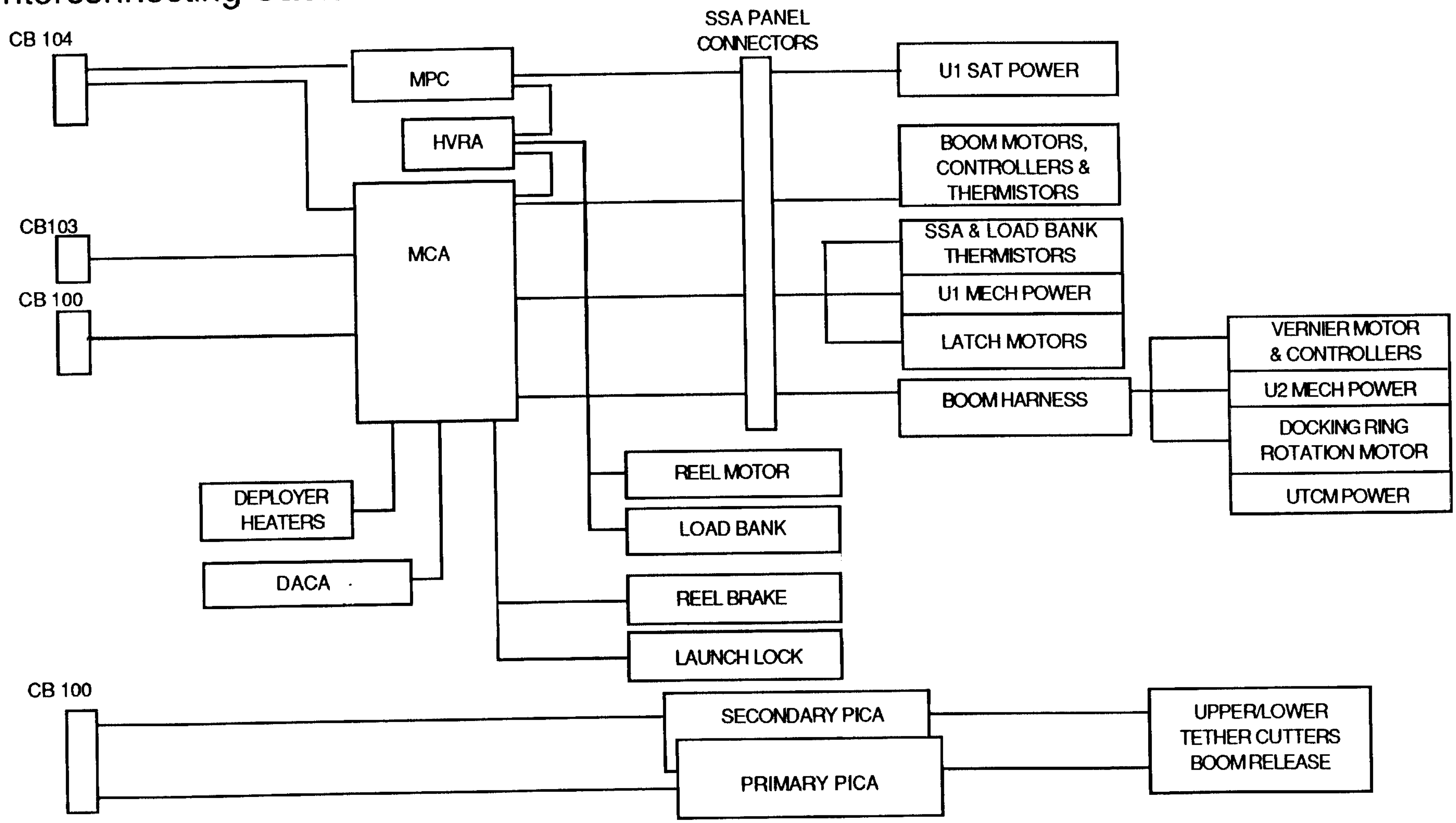
Pallet Mounted Cabling
Satellite Support Structure Cabling

The Harness has the Following Characteristics:

Total Number of Terminations	~ 4600
Total Number of Connectors	148
Total Harness Weight	204 lb (136 lb Pallet, 68 lb SSS)

Electrical Power Distribution System

Interconnecting Cable Harness



Motor Power Conditioner

Description

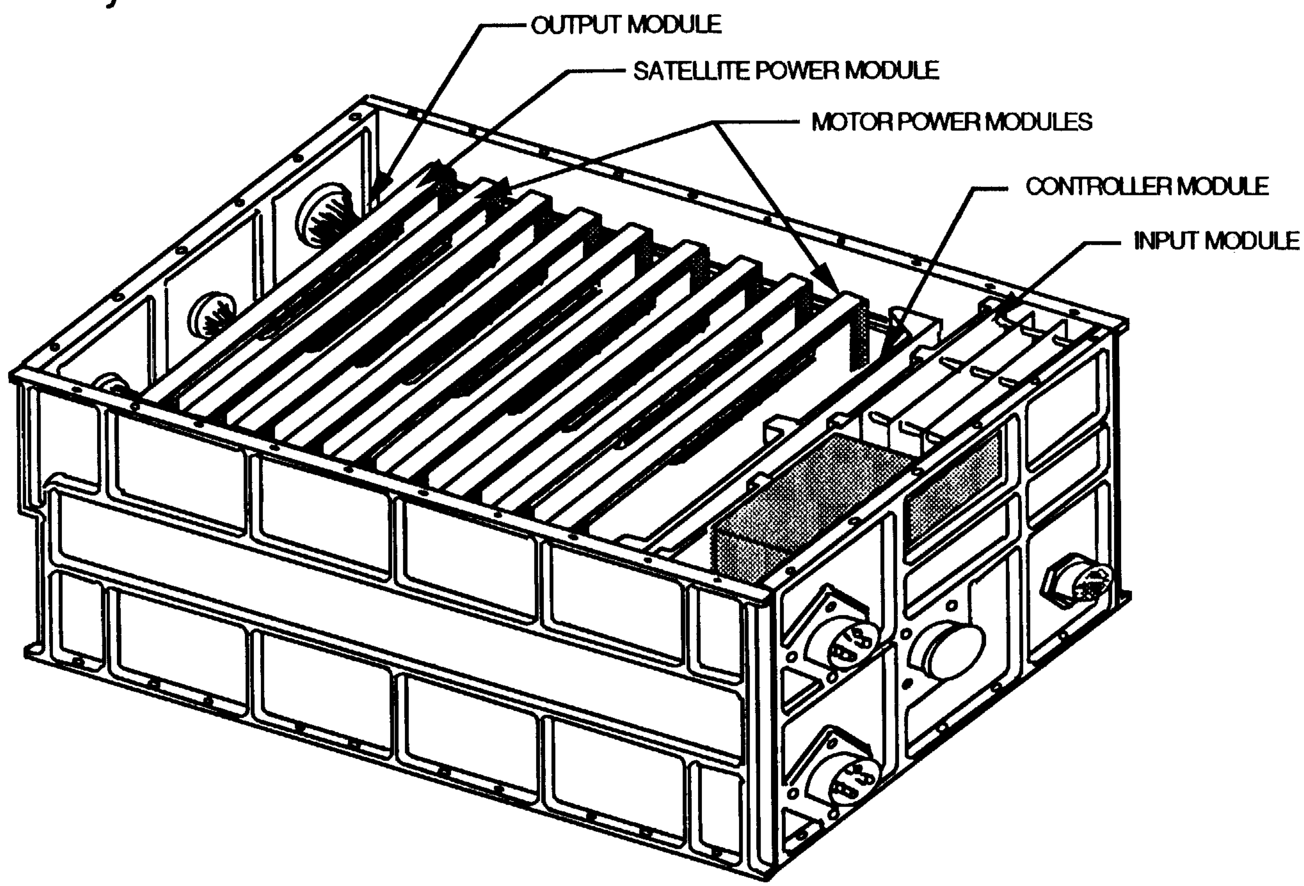
The Motor Power Conditioner (MPC) is a High Power DC-DC Voltage Convertor and Power Conditioner.

The MPC Performs the following functions:

Reel Motor Power Conversion -	2 Crew Selectable Output Voltage Levels, 120 ±6 Vdc 26 ±1.3 Vdc 25 A steady-state, 32 A peak
Controlled Voltage Slewing During Output Staging -	0.32 to 4.7 Volts/second
Satellite External Power Conversion -	33 ±3 Vdc 12 A steady-state
Orbiter Bus Isolation & EMI Control-	130 dB above 1 µA, 30 Hz to 2 kHz

Motor Power Conditioner

Top Assembly



Motor Power Conditioner

Hardware Description

The MPC Consists of 12 Modules
- 1 Input Module
- 1 Controller Module
- 1 Satellite Power Module
- 8 Reel Motor Power Modules
- 1 Output Module

The Input Module Contains an EMI Filter to Reduce Conducted Emissions, a Relay to Provide an Enable/Disable Function, and an Inrush Limiter to Limit the Input Current at Power Up.

The Controller Module Contains the Clock Generator Which Provides 8 Independent Clock Outputs to Control the Switching of the Power Modules Which Ensures Load Sharing. The Over Current Protection Senses Output Current and Compares it to a Selected Limit. The Clock is Shutdown if the Limit is Exceeded.

The MPC Uses 8 Parallel Power Modules with Local Current Control and an Overall Voltage Control Loop. Multiple Power Modules are Used to Achieve the 3 kw Power Level Required.

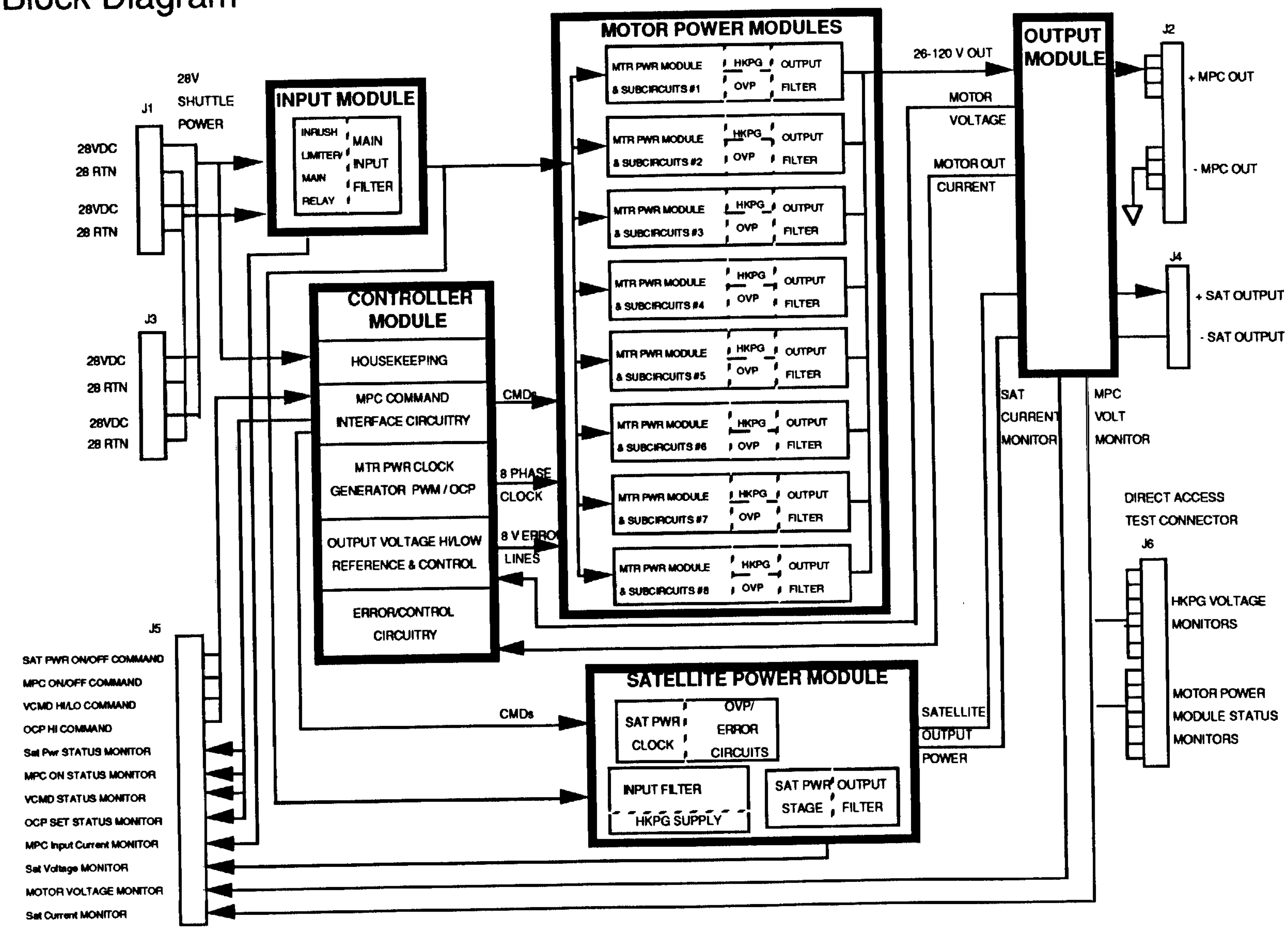
The Satellite Power Module Contains the Same Circuit Elements as the Motor Power Modules Plus an Independent Controller Card.

The Output Module Provides Output Fusing and Common Mode Filtering.

Telemetry Outputs Include Input Current (Low and High Range), Output Voltages (Motor and Satellite), Satellite Output Current, and Discrete Status Signals (Motor On/Off, Sat On/Off, VCmd Hi/Lo, OCP Set)

Motor Power Conditioner

Block Diagram



Motor Control Assembly

Overview

The Motor Control Assembly (MCA) Provides the Following Functions:

Reel Motor Power Modulation and Control
Deployer Mechanisms Control
Signal Conditioning for the Deployer Instrumentation
Deployer and Satellite Heater Control
Reel Brake Control

The MCA Receives +28 Vdc Orbiter Payload Bus A & B Power Through the Pallet Power Control Box (PCB). The PCB Contains Relays for Switching. The Following Relays are Utilized for the Deployer:

PCB K2	Deployer Heaters
PCB K3	MCA & DACA Power
PCB K4	Safety Bus #1 (Satellite Restraint Latches, Boom, Umbilical Retraction)
PCB K5	Satellite Heaters
PCB K6	Safety Bus #2 (Vernier Motor, Satellite Restraint Latches)
PCB K7	Satellite Heaters

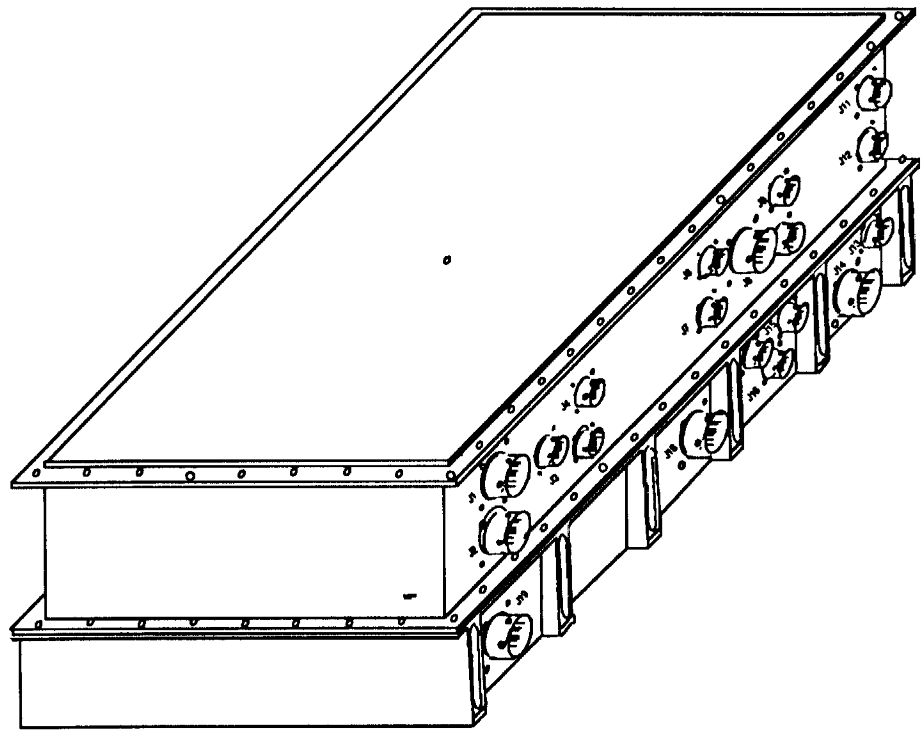
The MCA Also Receives Orbiter Auxiliary Bus Power Directly to Provide a 3rd Independent Method to Close the Satellite Restraint Latches.

Total MCA Housekeeping Power Consumption is 100 Watts Maximum.

Dimensions:	15.8 in. x 29.2 in. x 11.0 in
Weight:	130 lbs Maximum

Motor Control Assembly

Top Assembly



Motor Control Assembly

Lower Slice

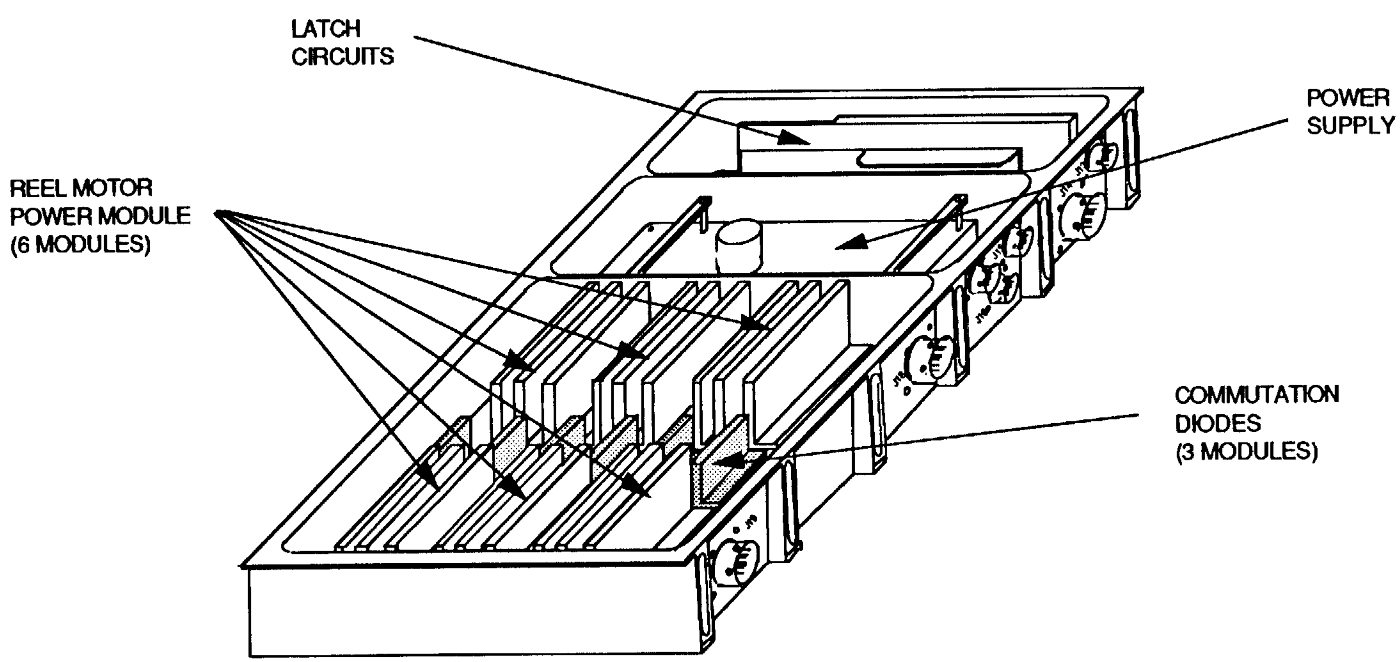
The MCA is Constructed as 2 Units, the Upper Slice and the Lower Slice.

The Lower Slice Contains the Housekeeping Power Supply, the Reel Motor Power Modules, Commutation Diodes, and Some of the Latch Circuits.

The Lower Slice is Connected to the Upper Slice By External Cables.

Motor Control Assembly

Lower Slice



Motor Control Assembly

Upper Slice

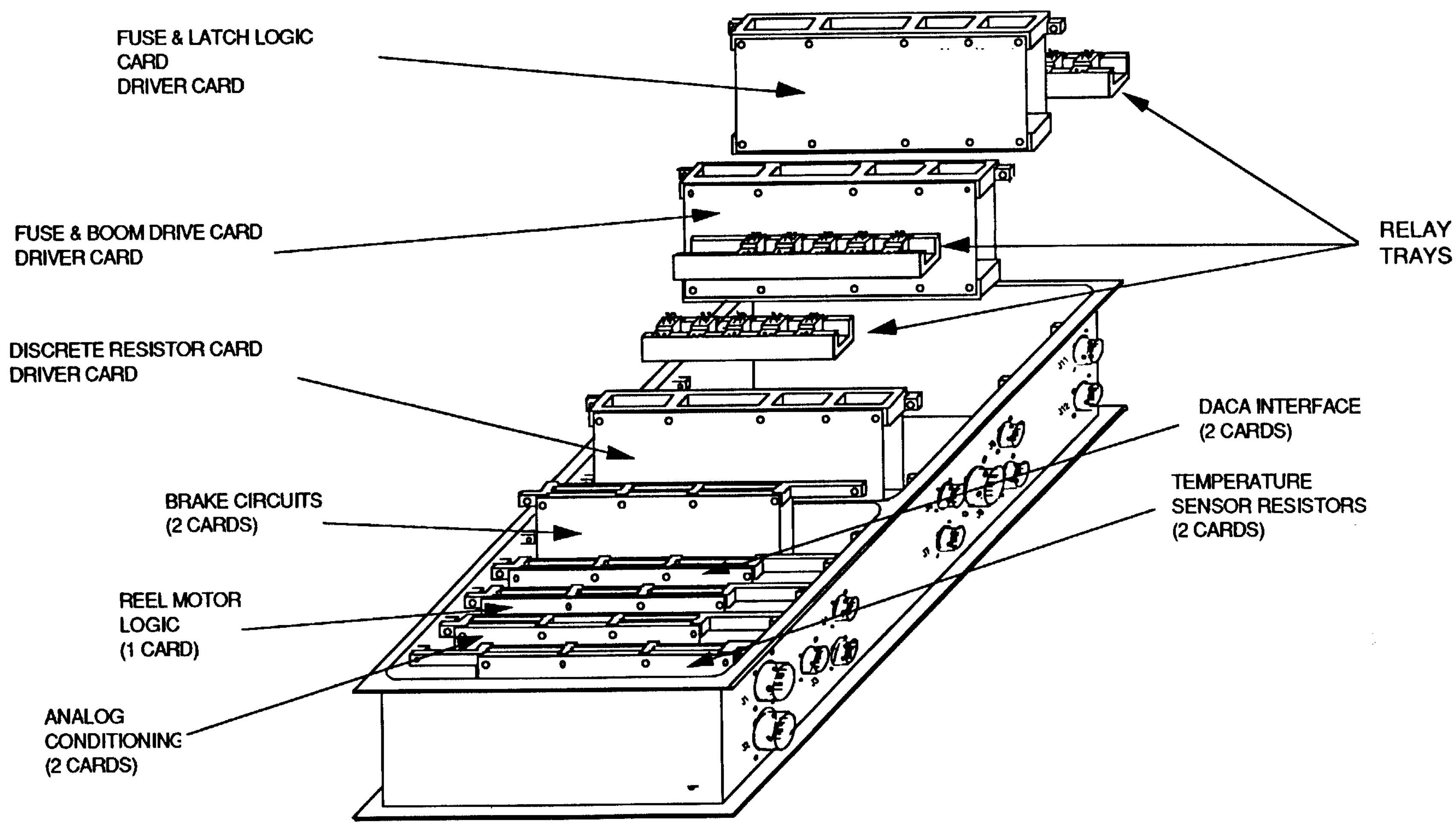
The Upper Slice of the MCA Contains the Circuitry for the Following Function:

- Reel Motor Control System
- Boom System
- Docking Ring System
- Umbilical Retraction System
- Launch Lock System
- Satellite Restraint Latch System
- Brake Application System
- Brake Off System
- Deployer Heater Power
- Satellite Discrete Commanding

Each of the Functions Listed Above Will be Discussed in Detail in the Following Pages.

Motor Control Assembly

Upper Slice



Motor Control Assembly

Reel Motor Control System

The Reel Motor Control System Provides Control of Reel Motor Torque by Pulse Width Modulation of the Reel Motor Current.

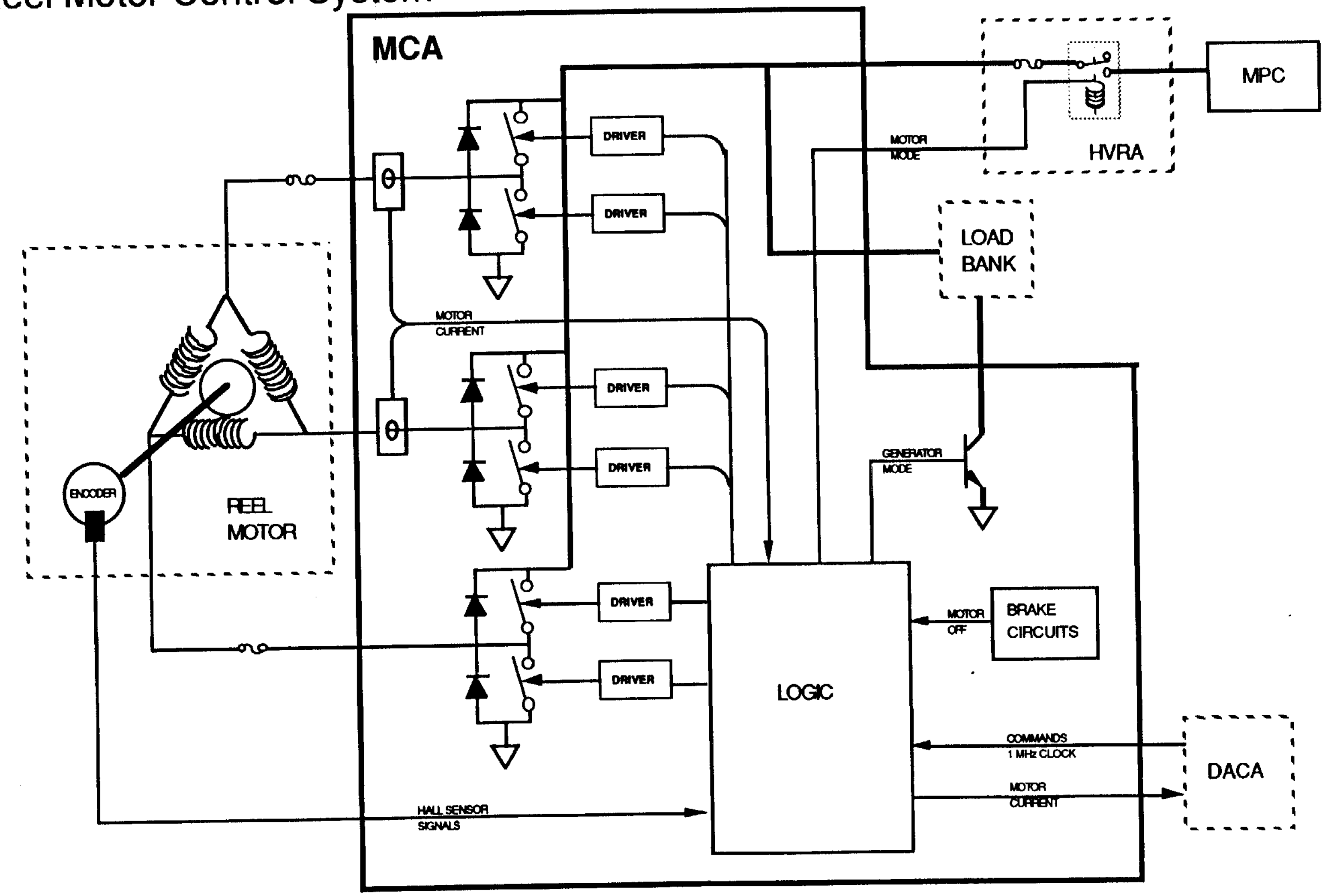
Power for the Brushless 3 Phase DC Motor is 26 V @ 6 A max for the 1st Mission and 120V @ 40 A max for the 2nd Mission.

The Pulse Rate is 1 kHz, with 511 Steps Available. The DACA Generates the Commanded Pulse Width Based on the Programmed Control Laws or Crew Inputs. The Commanded Pulse Width, Motor Current, Motor Shaft Position, and the Brake On/Off Discrete are all Utilized to Modulate the Current.

Protection Against Over Driving the Reel Motor is Provided by 15 Amp Fuses on the Motor, a 6 $\pm$1 Amp Motor Current Limit in the MCA, and the Brake On, Which Open the Motor Circuit.

Motor Control Assembly

Reel Motor Control System



Motor Control Assembly

Basic Operation (Motor Current Flow)

The Reel Motor Control System Operates the Reel Motor as a Motor to Retrieve the Satellite, and as a Generator to Provide Braking Torque During Satellite Deployment.

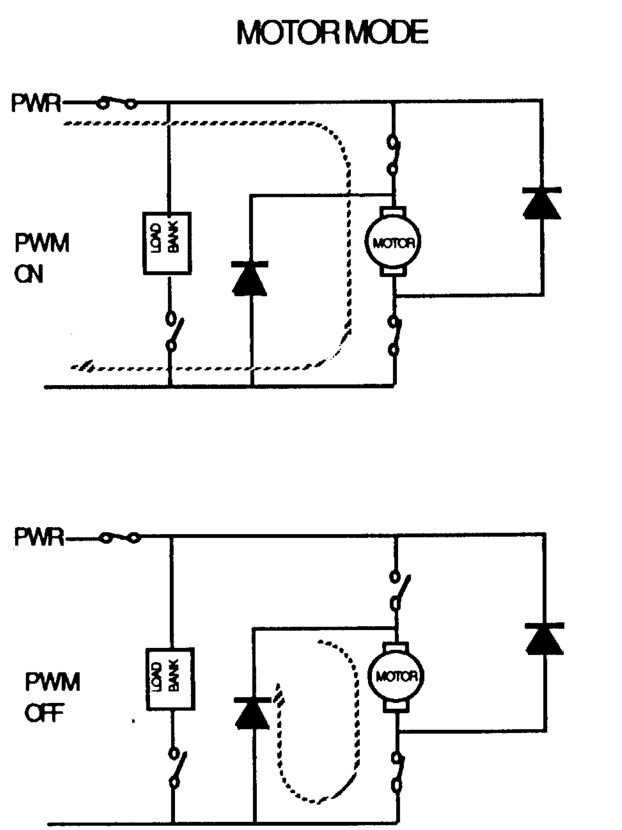
The Current Flow for the Motor and Generator Modes is Shown. In Both Cases, the Current Flow is Controlled by the DACA PWM Signal Which Drives the FET Switches in the MCA.

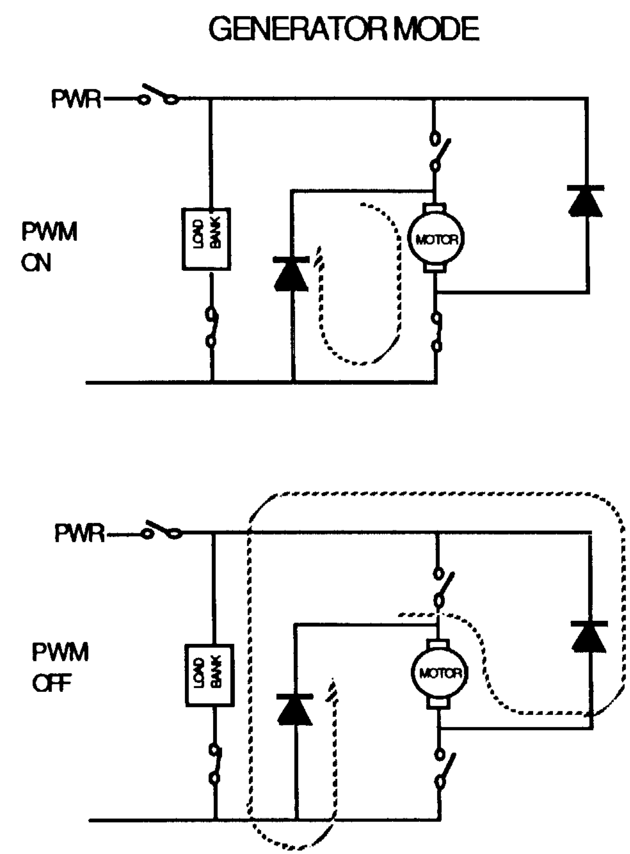
The Motor Current is Monitored by the DACA.

The Load Bank Resistance is 35Ω.

Motor Control Assembly

Basic Operation (Motor Current Flow)





Motor Control Assembly

Boom System

The MCA Provides 28 Vdc Power at 10 Amps Maximum to Raise and Lower the Satellite Deployment Boom. The Boom Motors are Brushless DC Types.

The Boom Stops Automatically at Full Extension, Full Retraction, and the Intermediate Positions. This is Accomplished by Removing Motor Power when the Microswitch Position Monitors Indicate that the Position has Been Reached.

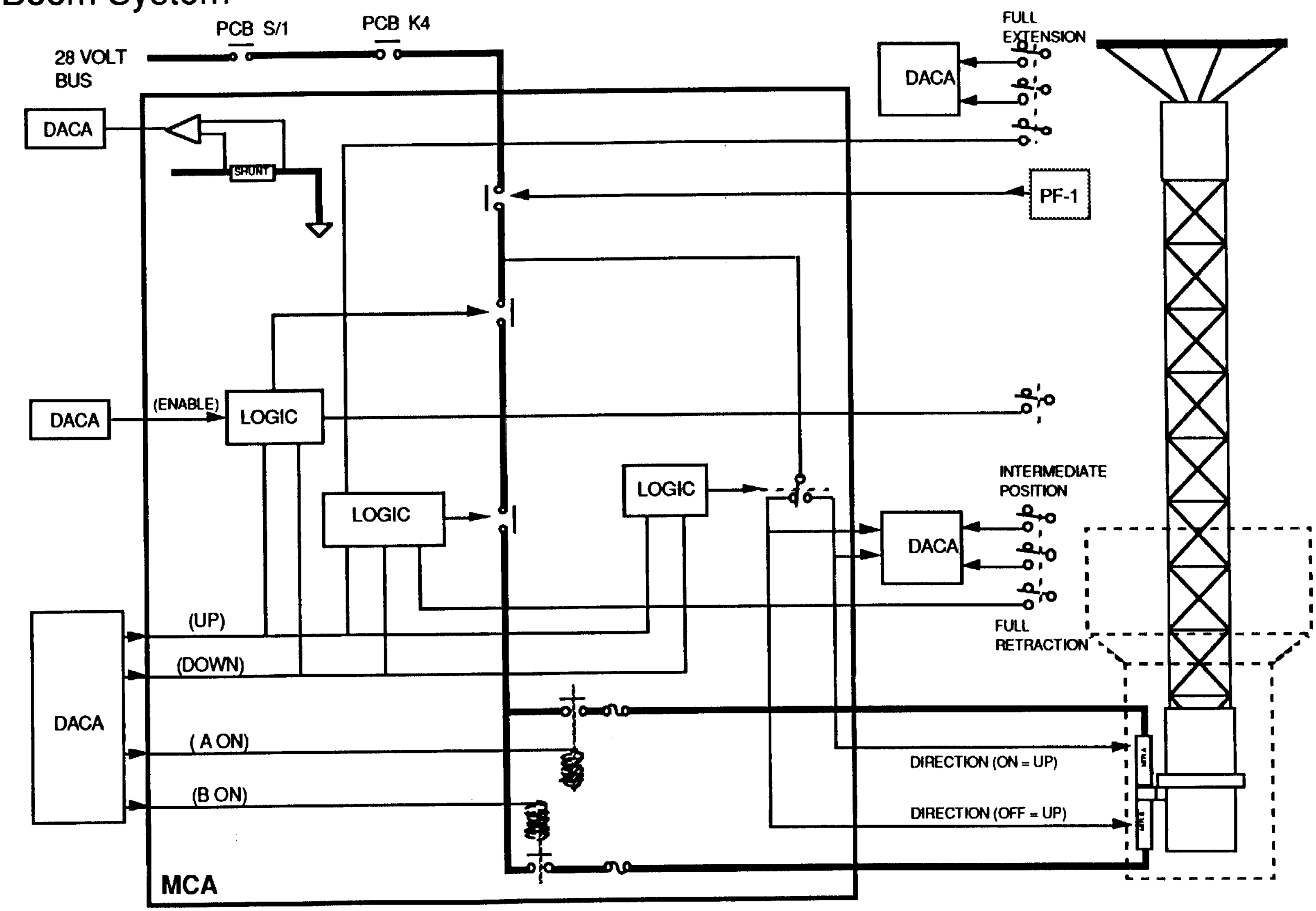
Redundant Microswitches Indicate Full Extension and Retraction Positions to the Crew and Ground Controllers. Motor Current and Up/Down Direction is Also Monitored.

The Boom Motors and Controllers are Redundant and Contain Heaters for Thermal Control.

2 Commands are Required to Extend or Retract the Boom. The Enable Command is Direct from the Orbiter PF-1MDM, and the Up/Down Command is from the DACA. The Motor Selection, Direction, and Intermediate Stop Functions are Commanded from the DACA. The Power Removal Function is Redundant.

Motor Control Assembly

Boom System



Motor Control Assembly

Docking Ring System

The MCA Provides 28 Vdc Power at 1 Amps Maximum to Rotate the Docking Ring for Satellite Alignment.

The Docking Ring Rotates ±185° and is Driven by a Brush Type Motor. Mechanical Stops Limit Docking Ring Rotation.

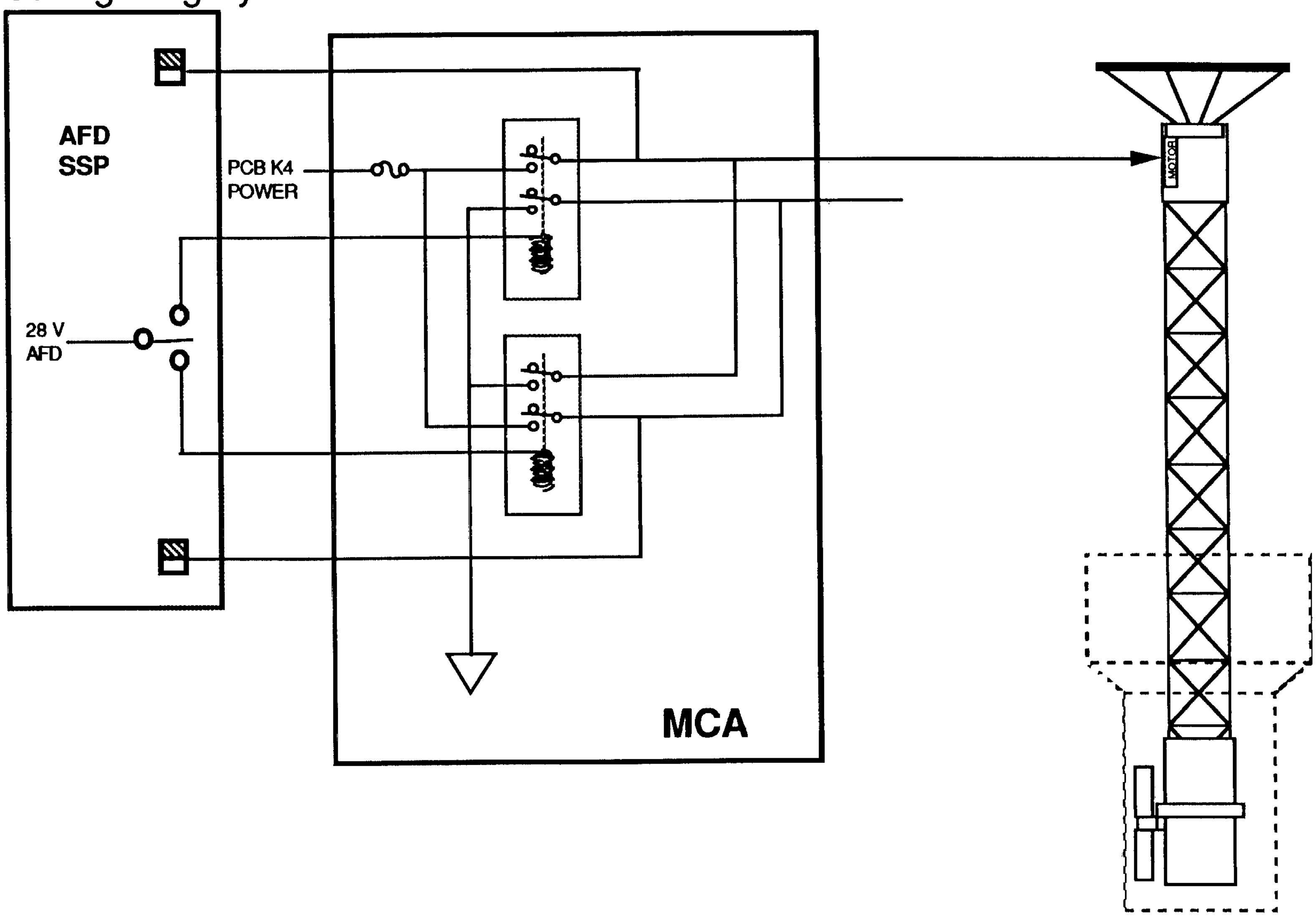
The Docking Ring is Commanded by the Orbiter Standard Switch Panel (SSP).

The SSP Receives a Talkback to Indicate Rotation Direction.

The Motor is Heated for Thermal Control, and the Motor is Fused with a 10 Amp Fuse.

Motor Control Assembly

Docking Ring System



Motor Control Assembly

Umbilical Retraction System

The MCA Provides 28 Vdc Power at 2 Amps Maximum to the Umbilical Mechanism's Brush type DC Motors to Disconnect, Retract, and Retain the U1 and U2 Umbilicals.

The U1 Motor Stops Automatically at Full Retraction by Removing Motor Power when a Microswitch Position Monitor Indicates that Full Retraction has Been Reached.

The U2 Motor Stalls at Full Retraction. Motor Power is Removed after a Fixed Timeout Period.

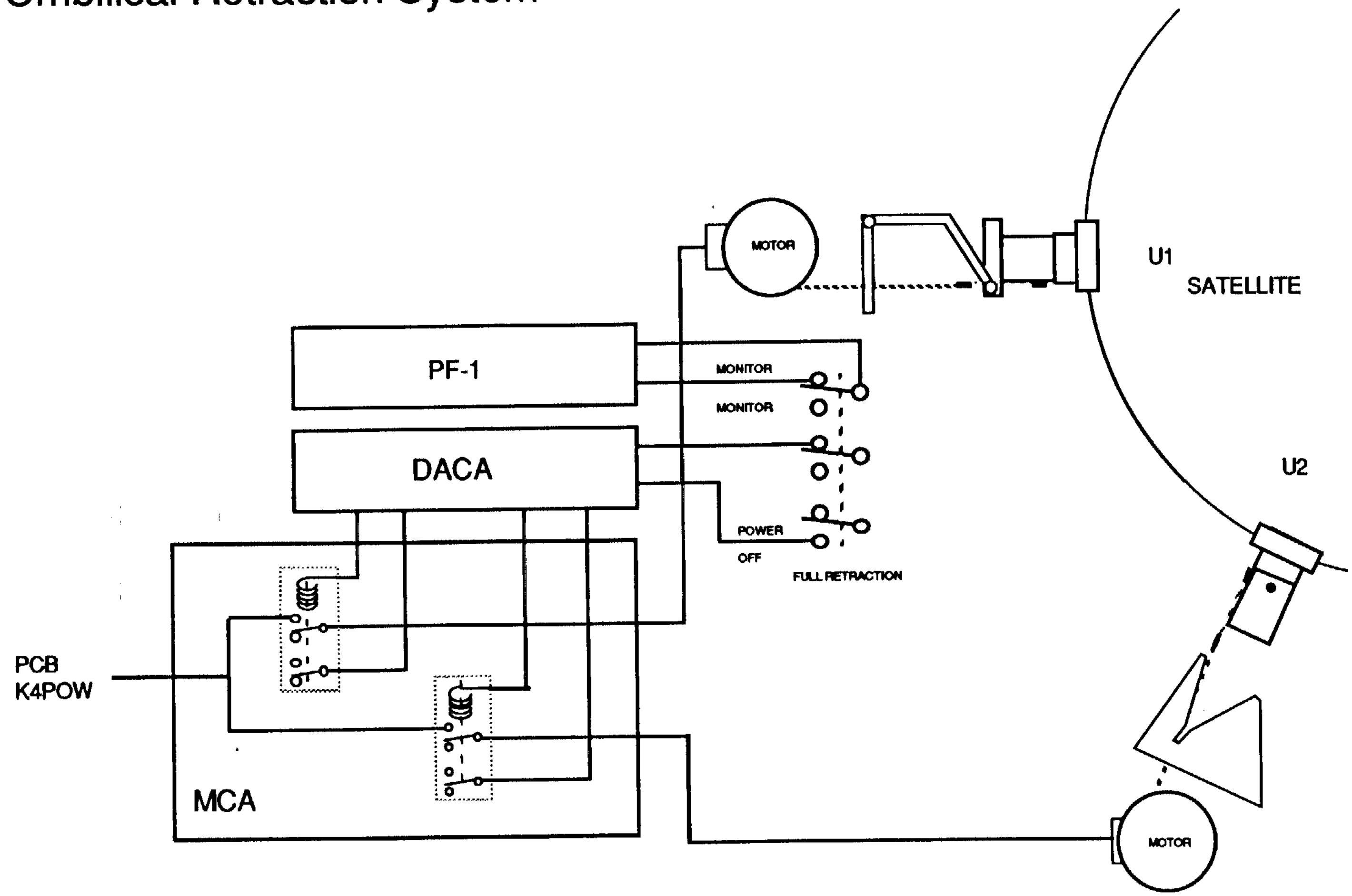
Triple Redundant Microswitches Indicate U1 Fully Retracted to the Crew and Ground Controllers. U1 and U2 Separation and the Control Relay Position is Monitored.

The Commands to Retract the Umbilicals are Received from the DACA.

The Motors are Fused and have Heaters for Thermal Control.

Motor Control Assembly

Umbilical Retraction System



Motor Control Assembly

Launch Lock Disengage System

The Launch Lock Disengage System Controls the Motor that Disengages the Launch Lock. The Motor is a Brush Type, 28 Vdc Motor that Draws 1 Amp Maximum.

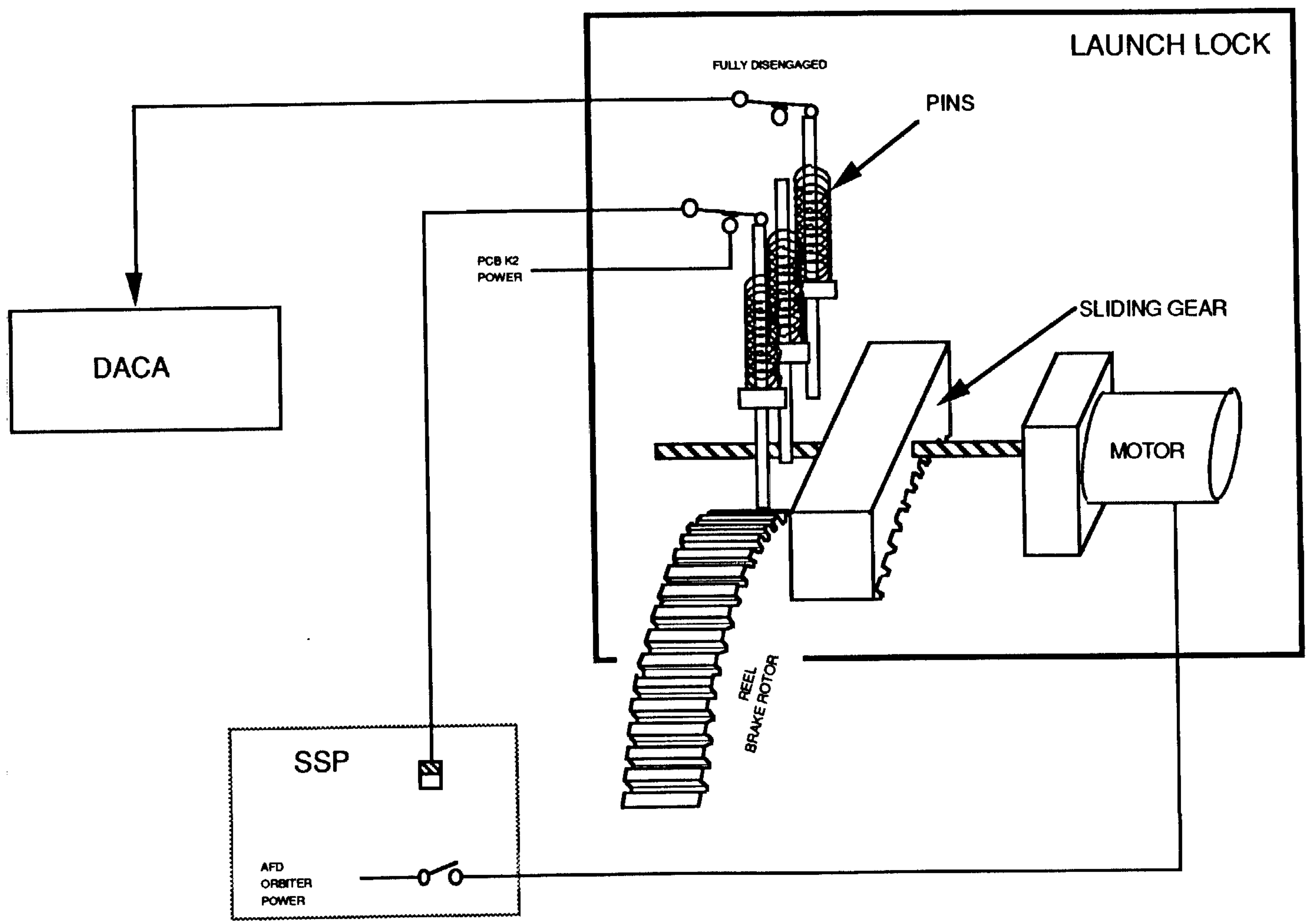
The Launch Lock Disengage Function is Crew Commanded by a Momentary Contact Switch on the SSP.

2 Switches Provide Feedback that the Lock is Fully Disengaged, 1 to the DACA for Telemetry, and 1 to the SSP. The Power is Removed from the Launch Lock Motor when the Crew Releases the Switch.

The 3 Spring Loaded Holdback Pins Prevent Re-engagement.

The Motor has a Heater for Thermal Control.

Motor Control Assembly
Launch Lock Disengage System



Motor Control Assembly

Satellite Restraint Latch System

The Satellite Restraint Latch (SRL) System Controls the Opening and Closing of the SRLs.

A Single Path is Provided to Open the Latches. Opening the Latches Requires 2 Independent Commands from Separate Sources and the Application of Power to Prevent Inadvertent Opening. Power is Removed During Launch and Reentry.

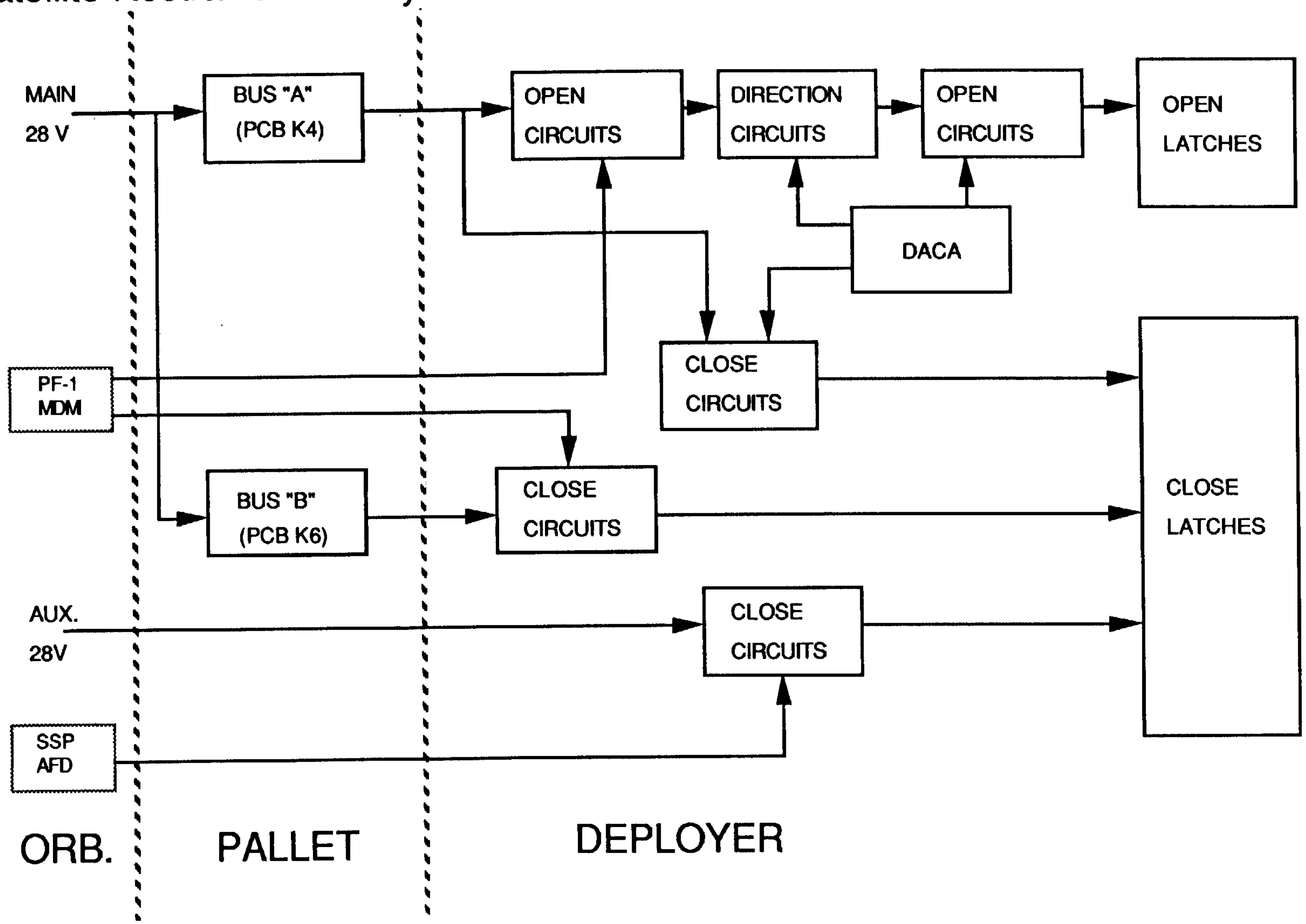
3 Independent Paths are Available to Close the Latches. Both the Power and the Commands are Independent.

To Open the Latches the Latch Open Enable is Sent by the Orbiter PF-1MDM. The DACA Controls the Direction Circuits and the Open Latch Command. The Latches Open and Close in 2 Groups of 3. All 6 Latches Must Open to Deploy the Satellite.

There are 3 Independent Methods to Close the Latches. The First Method Utilizes the Payload Bus "A" Power and the Close Command from the DACA. The Second Method Utilizes Payload Bus "B" Power and the Close Command from the Orbiter PF-1 MDM. The Third Method Utilizes Auxiliary Bus Power and a Close Command from the Orbiter SSP.

Motor Control Assembly

Satellite Restraint Latch System



Motor Control Assembly

Satellite Restraint Latch System

Each Latch Requires 28 Vdc Power at 1 Amp (Nominal), 5 Amps (Stall).

The Fully Open, Fully Closed, Direction, and Group Motor Current is Monitored.

Each Latch Contains 6 Limit Switches, 1 to Provide Automatic Shutoff when the Latch is Open, 1 to Provide Automatic Shutoff when the Latch is Closed, 2 to Indicate Fully Open, and 2 to Indicate Fully Closed.

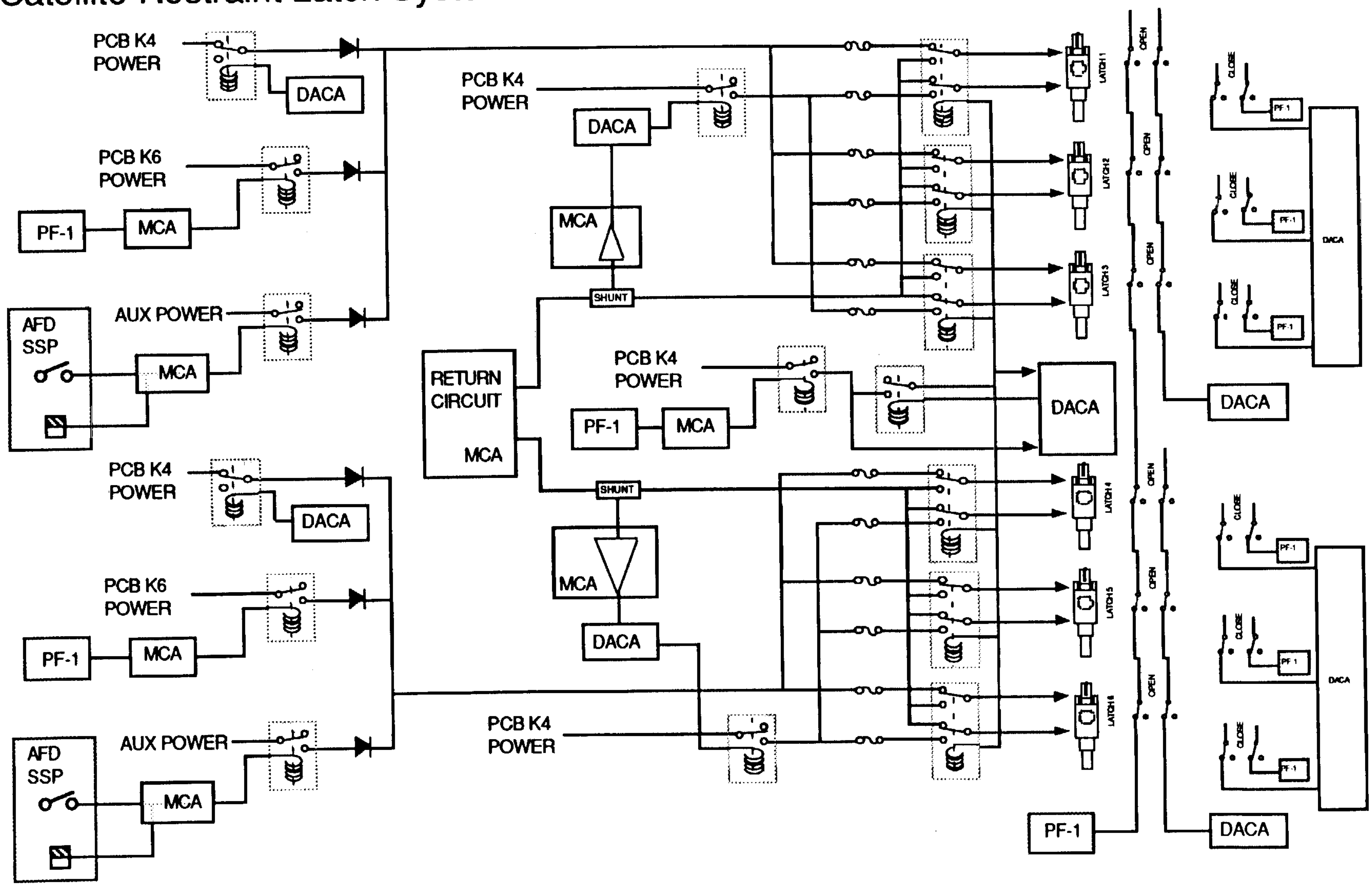
Latch Open Monitors Include an All Latches Open Discrete to the PF-1 MDM and 2 Latch Group Opens to the DACA.

Latch Closed Monitors Include Individual Latch Closed Discretes to Both the PF-1 MDM and the DACA. To be Secured for Reentry, any 4 Latches must be Verified Closed, or any Group of 3 Must be Closed.

The Latch Circuit Relays are Normally Closed to Allow Latch Closure Under Fault Conditions.

Motor Control Assembly

Satellite Restraint Latch System



Motor Control Assembly

Brake Application System

The Brake Application System Prevents Run-away Tether Situations that Could Result from Motor and/or Control System Failures.

The Brake Will Automatically Apply Itself if any of the Following Situations Occur:

1. Tether Rates Exceed 0.1 m/s or 0.4 m/s (Limit is Crew Selectable).
2. Tether Length Changes More Than ±180 Meters.
3. System Power is Lost.

The Brake System Supplies 28 Vdc Power to the Hold-Off Brake to Prevent the Spring Loaded Reel Brake From Applying. When This Power is Removed, the Reel Brake Automatically Applies. There are 2 Separate Circuits to Supply this Power to the Brake. The Brake System is Fail Safe, with No Control by the DACA.

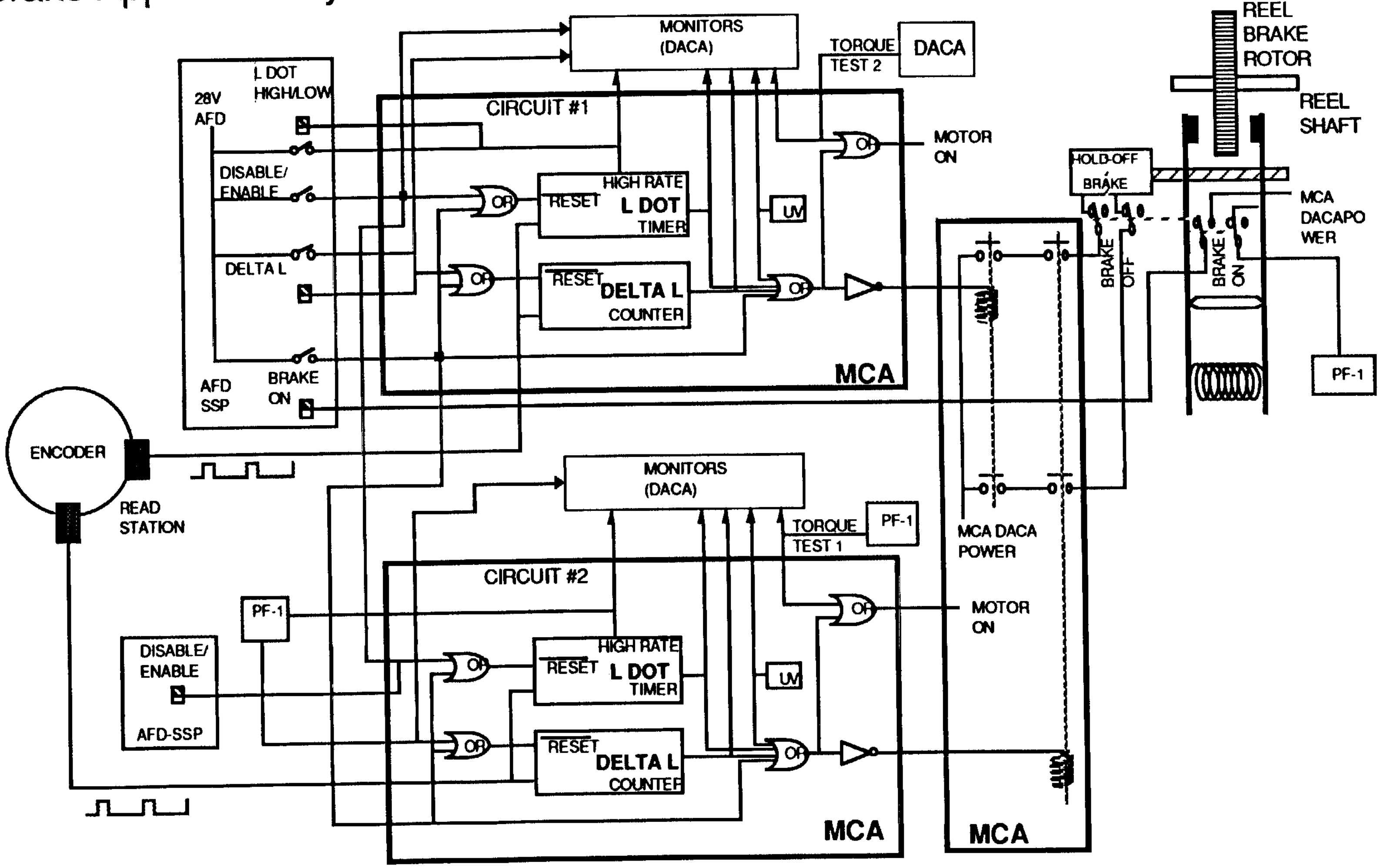
The Brake Application System Employs 2 Separate Encoder Read Stations and Circuitry to Read Tether Length and Rate. If an Out of Limit Condition Exists on Either Circuit, Power is Removed from the Hold-Off Brake, and the Reel Brake is Applied. The Brake Can be Reset Only by Setting the SSP Brake Switch to ON.

Commanding is by Crew Action Only, Via the SSP and the PF-1 MDM. Commands include High/Low Rate Selection, Rate Disable, Length Enable, Brake ON/OFF, and Torque Test. The Torque Test Allows Reel Motor Torque While the Brake is On.

System Monitors Include Status of Each Circuit to the DACA, and Brake Closed Discretes to the PF-1 MDM and SSP. The Brake Closed Discretes are Implemented with Redundant Switches.

Motor Control Assembly

Brake Application System



Motor Control Assembly

Brake Off System

The Brake Off System Removes the Reel Brake Torque by Driving the Brake Calipers Apart.

The Brake System Supplies 28 Vdc Power to Drive the Motor Which Opens the Reel Brake Calipers. When the Brake is Fully Open a Limit Switch is Reached, and the Hold-Off Brake is Then Energized. After a 3 Second Delay, Power is Removed from the Motor

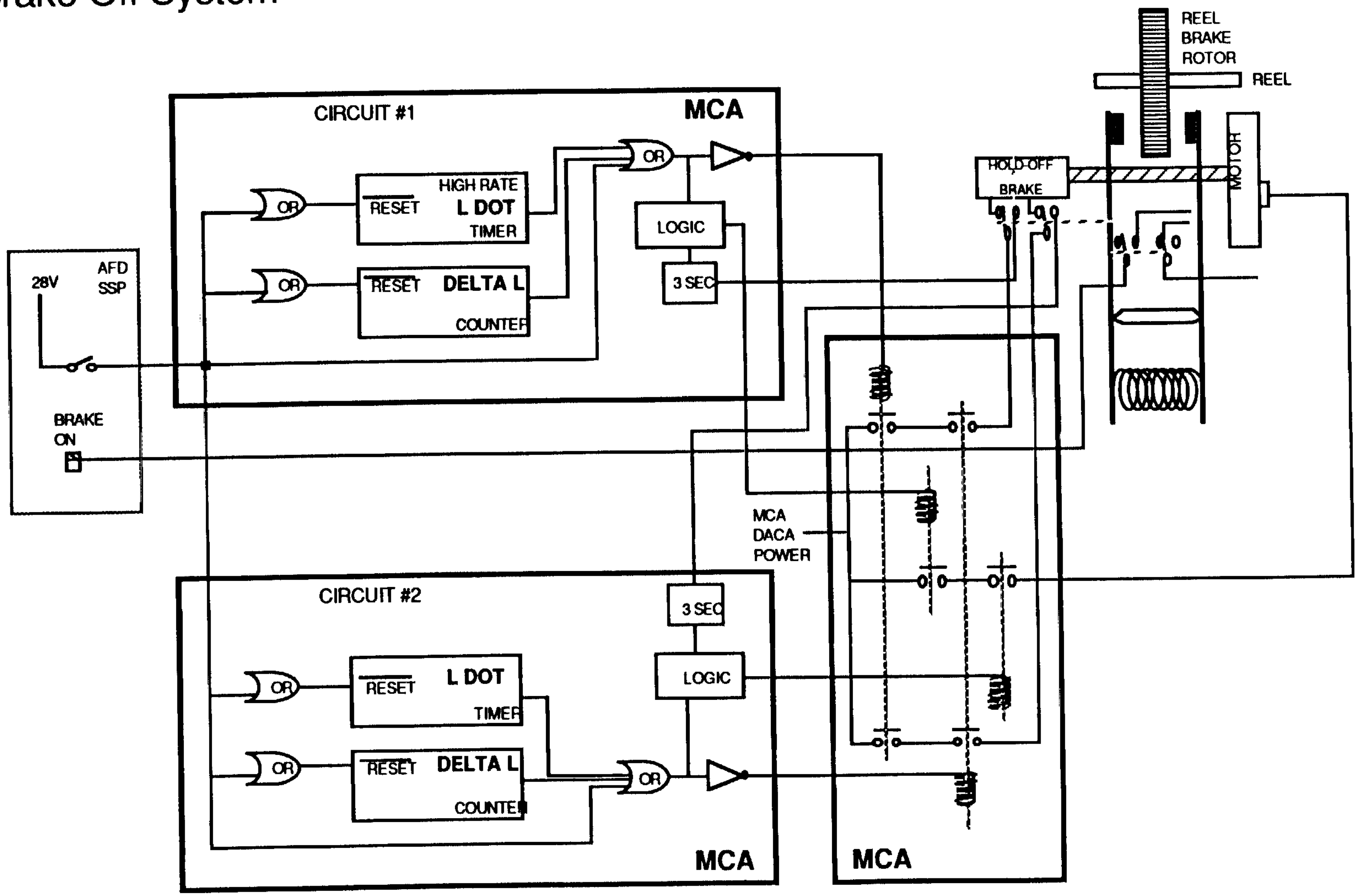
Commanding is by Crew Action Only, Via the SSP Brake ON/OFF Switch.

The System is Fail Safe, as Loss of Power Removes the Motor Power and the Brake Re-Applies.

The Brake Motor is Fuse Protected.

Motor Control Assembly

Brake Off System



High Voltage Relay Assembly

Description

The HVRA Provides the Following Functions:

- Isolation Relay Between the MPC and the MCA to Prevent Inadvertent Reel Motor Powering.

Relay Description:	Single Pole, Single Throw, 80 A, 326 V
Coil Voltage:	33.6 V (max), 16.0 V (pickup), 7.1 V (hold)
Coil Current:	3.73 A (max inrush), 0.38 A (max hold)

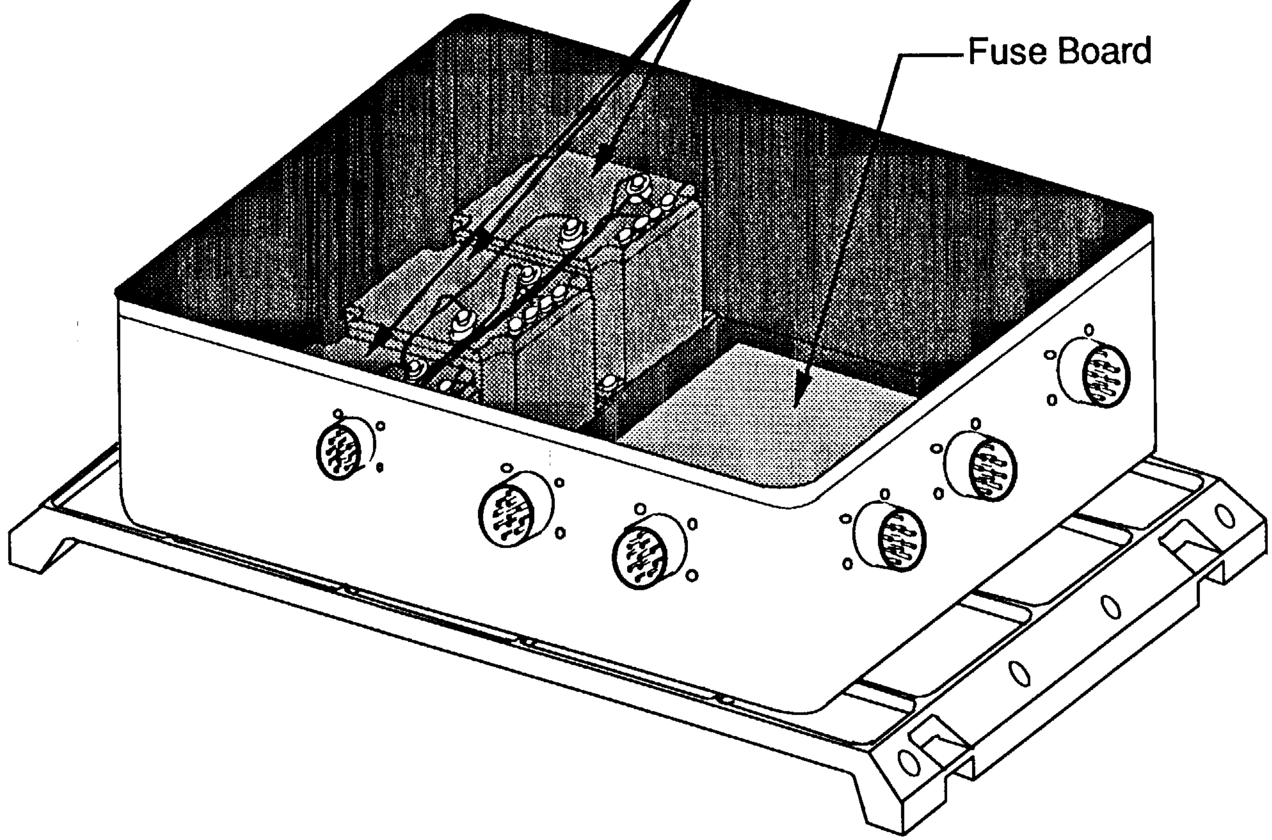
- 15 Amp Fuses to Current Limit Reel Motor Power.
- Power Switching Relays for the Load Banks During Satellite Deployment Operations when the Reel Motor is Operating in Generator Mode (Not Used on TSS-1).

The HVRA Contains Heaters with Series Redundant Thermostats to Provide Active Thermal Control.

Dimensions:	14.5 in. x 28.2 in. x 7 in.
Weight:	45 lbs

High Voltage Relay Assembly

Top Assembly



Satellite Interface

Overview

The Deployer Supplies Power and Commands to, and Monitors the Satellite Prior to Deployment.

The Following Power is Supplied:

Heater Power	28 Vdc, 10 A max
Checkout Power	33 Vdc, 12 A max

The Satellite Command Interface Includes the Following:

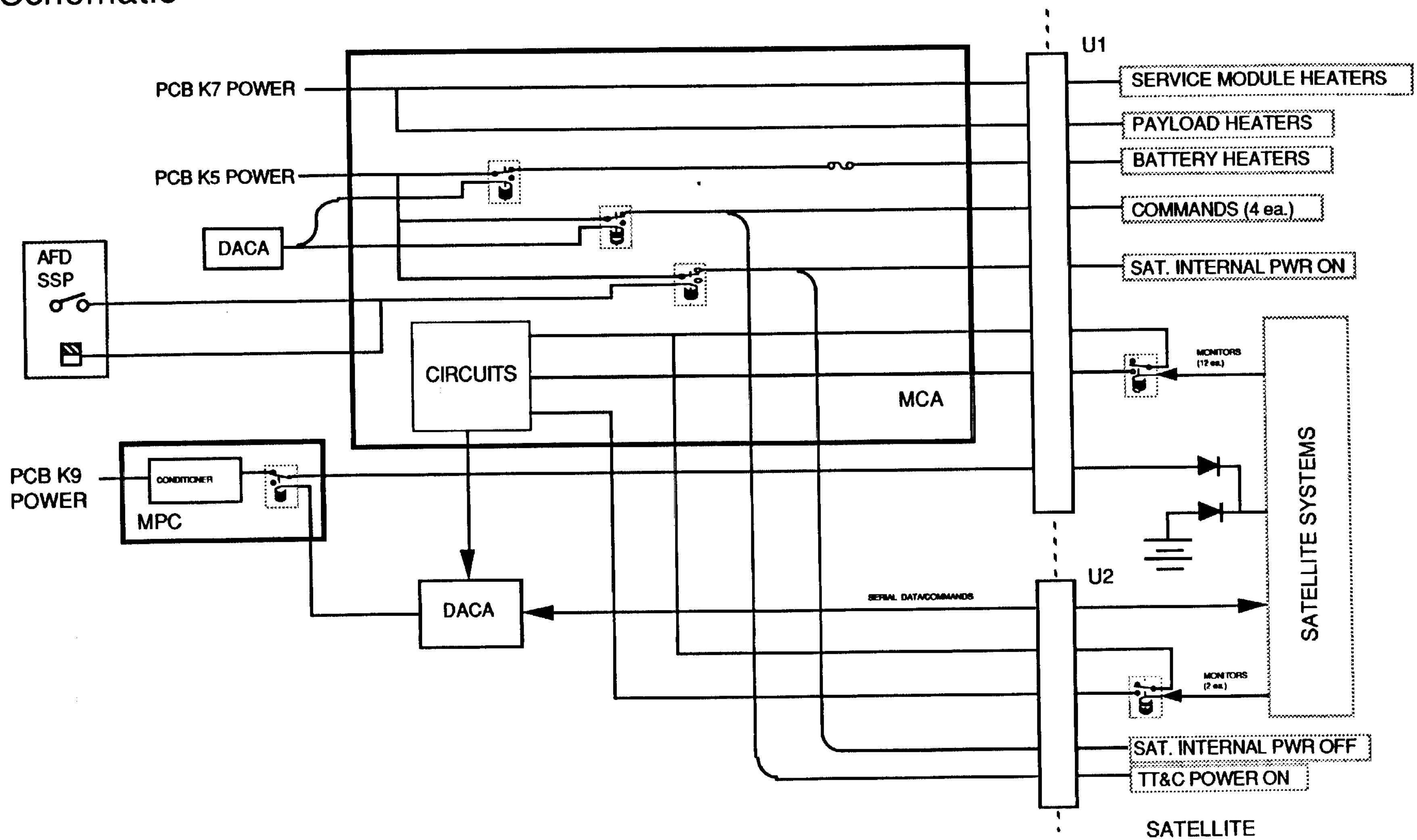
- Serial Commands From the DACA Through U2
- 28 Vdc Discrete Commands Through U1
- Standard Switch Panel Discrete Commands
- DACA Discrete Commands

The Satellite Monitors Include the Following:

- 12 Contact Closure Monitors
- Serial Data to the DACA
- Heater Power Current

Satellite Interface

Schematic



Command & Data Management

DACA

The DACA Provides Control and Monitoring of the Deployer Systems.

The DACA Computational Components Consist of an 80C86 CPU with a 8087 Numeric Coprocessor, Operating with a 5 MHz Clock Rate. The 69K 8 Bit Word Program is Stored in the DACAs 128K 8 Bit Word EPROM. There a is 16K 8 Bit Word RAM.

The DACA Input/Output Capability is as Follows:

Analog	32 Differential Channels 64 Single Ended Channels
Discrete Inputs	128 Inputs
Discrete Outputs	64 Outputs
Relay Drivers	64 outputs
Clock Output	1 MHz
Serial Inputs/Outputs	8 Channels
GMT/MET	1 Input
SFMDM Interface	1 Input/Output
Telemetry Output	1 Output
Satellite Interface	Satellite Data at 64 or 16 KBS
Tether Measurement	Input From Encoder, 0.005 m/Pulse Tether Rate and Length Derived From Encoder Input

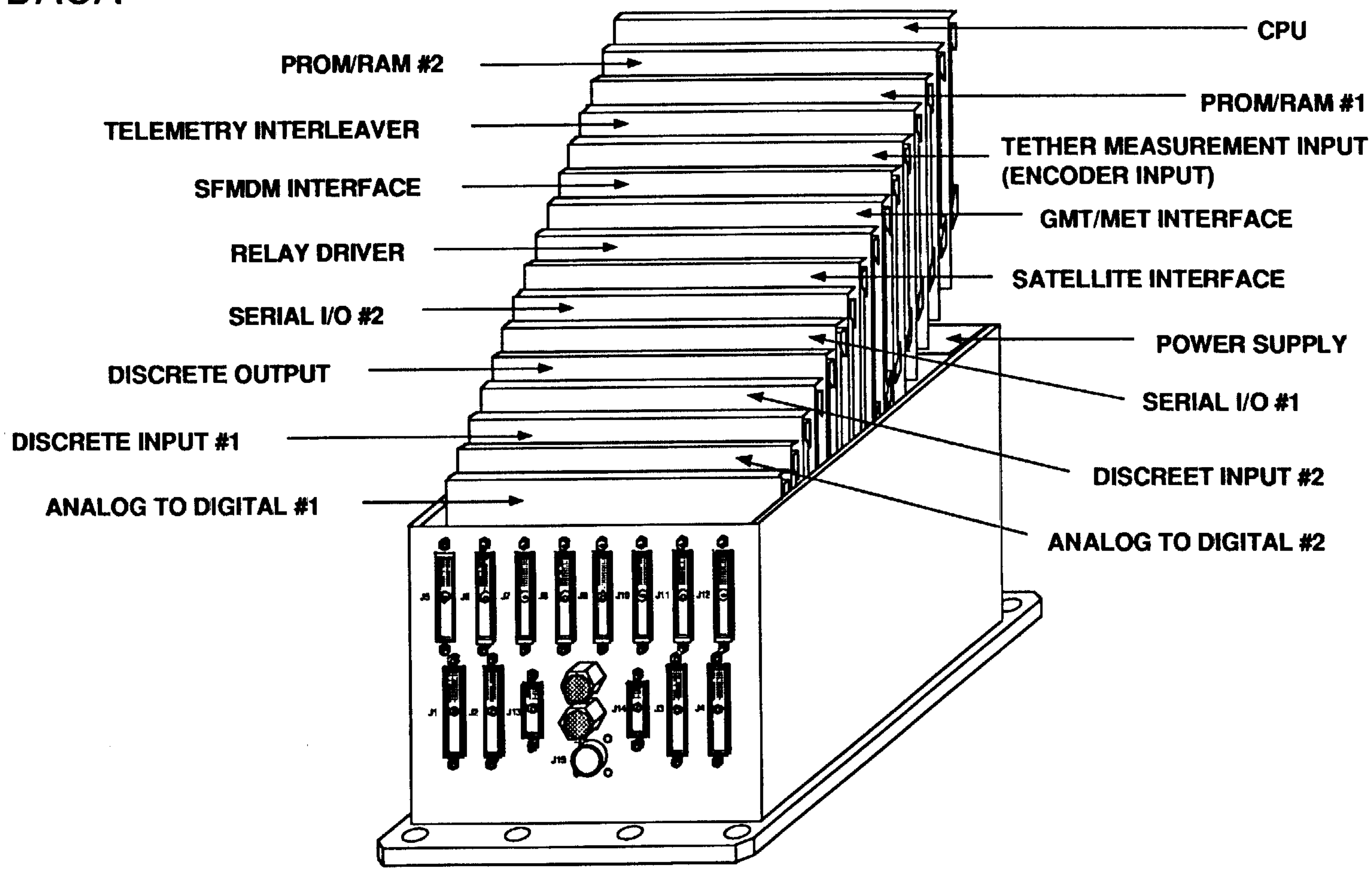
Total DACA Power Consumption is 100 Watts Maximum. It is Coldplate Mounted.

Dimensions: 8.83 in. x 11.58 in. x 7.0 in

Weight: 27 lbs Maximum

Command & Data Management

DACA



Command & Data Management

Command & Data System

The Command & Data System Gathers Data from Equipment and Assembles a Telemetry Stream to be Transmitted to the Orbiter. It Also Accepts and Processes Commands from the SFMDM.

The Measurements Collected by the Command & Data System Include the Following:

- 44 Temperatures
- 10 Analog Pyro Capacitor Bank Voltages
- 3 Tensiometers
- 8 Currents
- 34 Discrete Monitors
- Tether Length & Rate
- MCA Serial Status

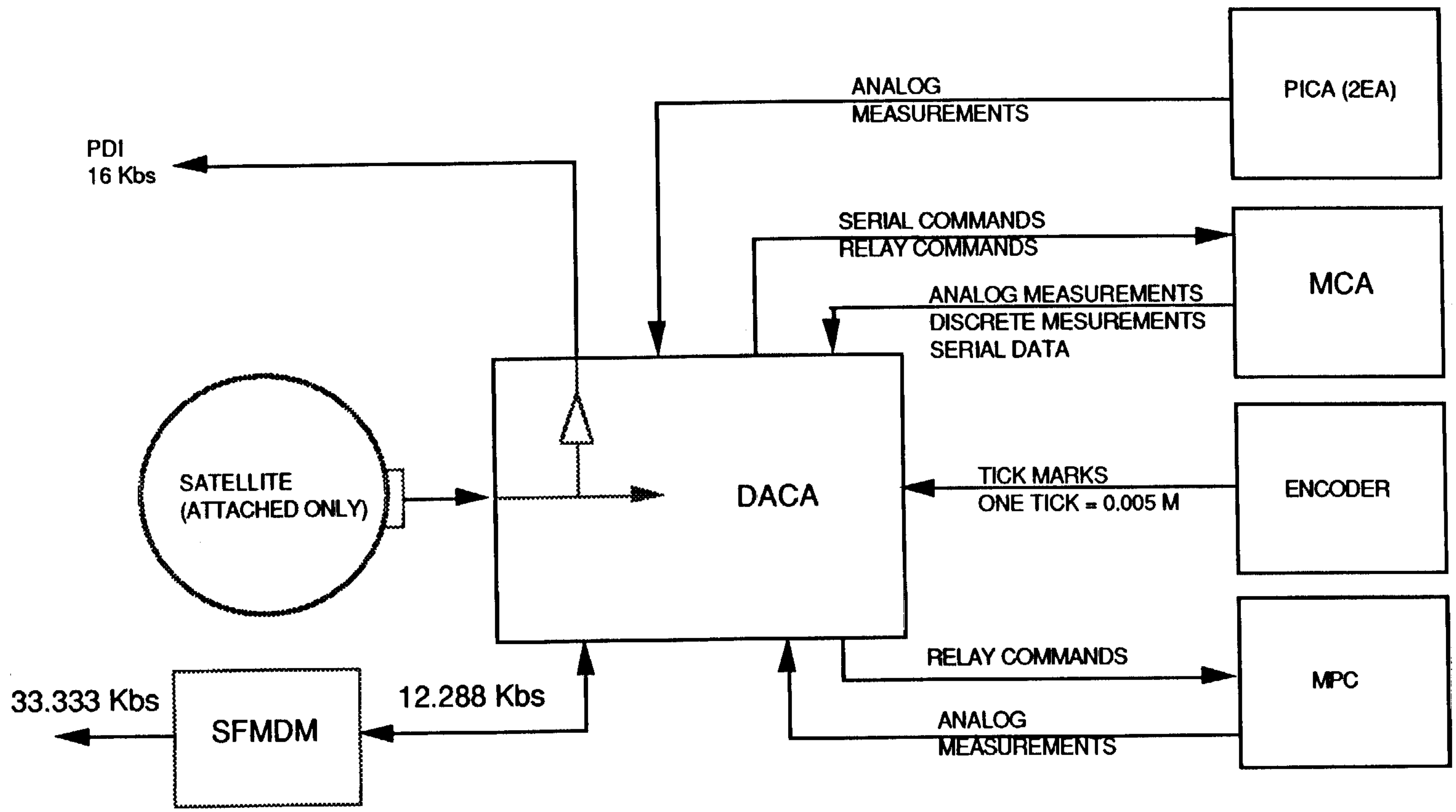
The Commands Received and Processed by the Command &Data System Include 37 Nominal Commands and 12 Contingency Commands.

Command and Data Inputs are Sampled Every 100 mSec, with a Processing Time of 100 mSec.

No Safety Critical Functions Utilize the Command & Data System.

Command & Data Management

Command & Data System



Command & Data Management

Tether Control System

The Tether Control System is a Closed Loop Feedback System Which Utilizes Tether Length and Rate Feedback to Drive the Reel Motor.

The Control System Utilizes Pulse Width Control to Operate the 3 Phase Brushless DC Motor as a Motor or Generator. A Vernier Motor is Used to Provide an Additional 30 N of Tension to Overcome System Friction During Deployment.

Feedback Data Includes Tether Length and Rate from the Encoder. Inboard and Outboard Tension, Motor Current, Supply Voltage and Current, and Supply Input Current data is Available but is not Used in the Control Loop.

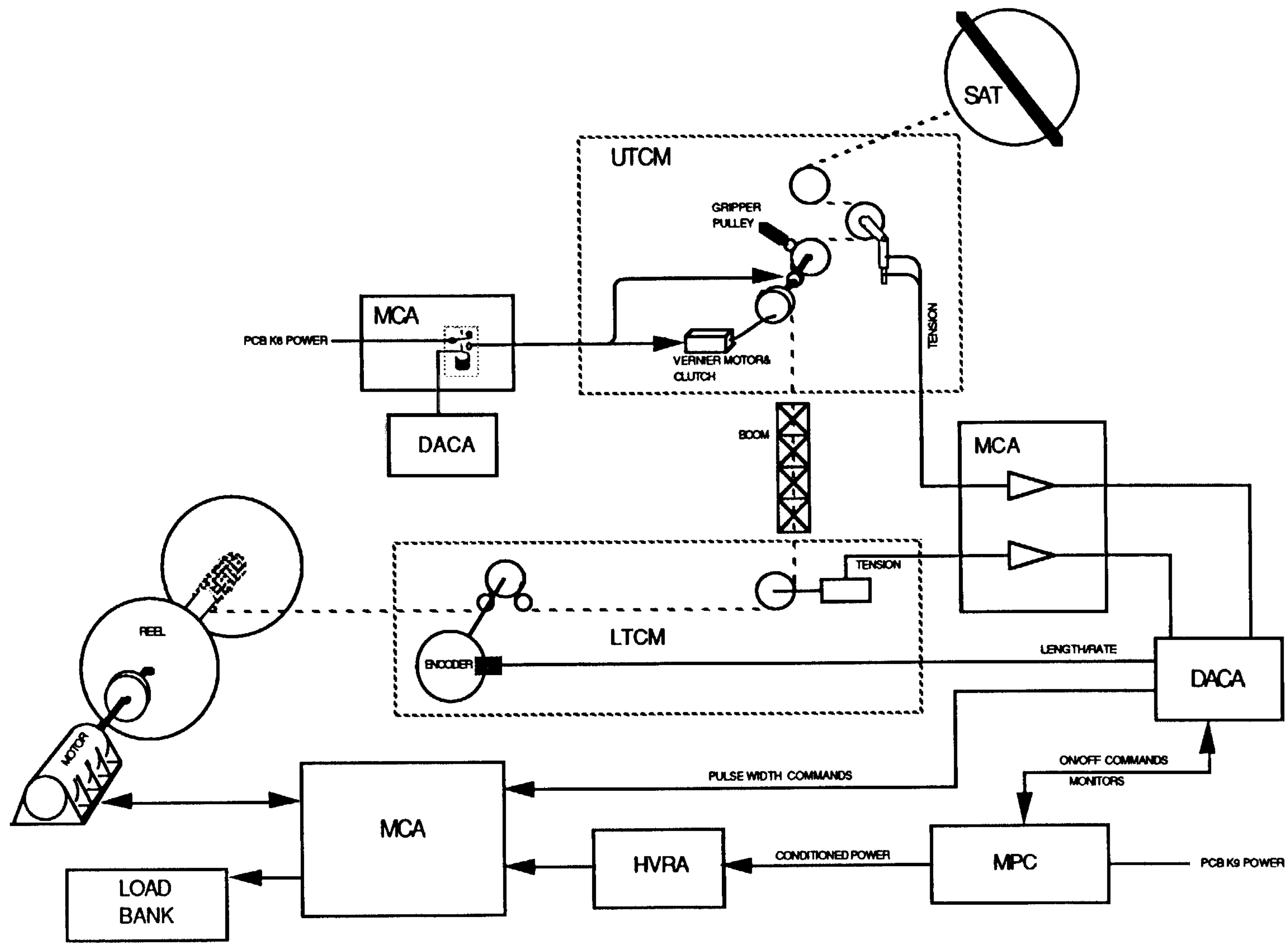
Overall System Rates and Tensions are Listed:

Deployment:	≈ 2.5 m/s max
Retrieval:	≈ 1.5 m/s max
Tension:	50 Newtons max

Control Loop Runaway Protection is Provided by the Reel Brake and Multiple Current Limits and Fuses.

Command & Data Management

Tether Control System



Deployer Flight Software

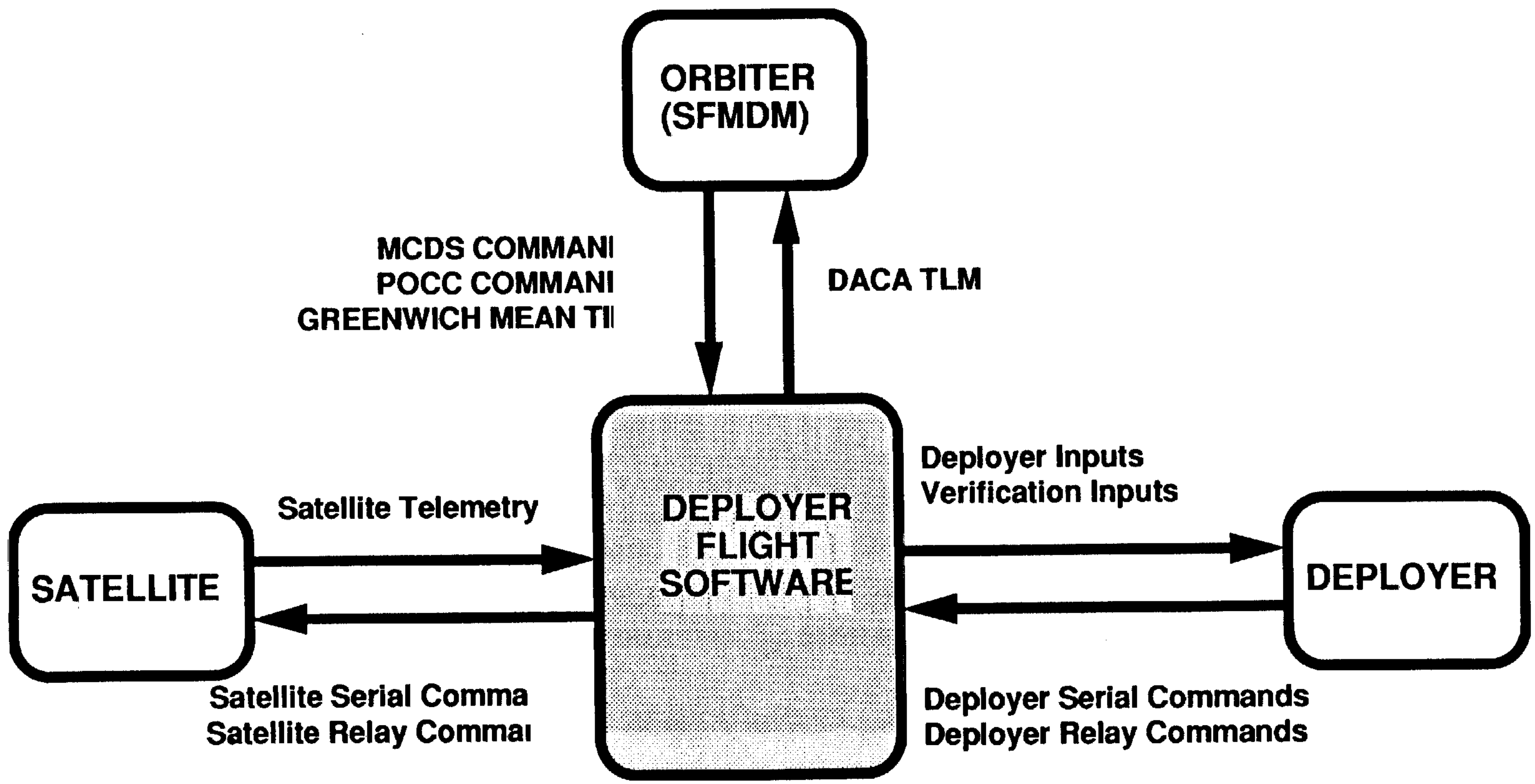
Top Level Requirements

The Deployer Flight Software is Designed to Perform the Following 6 Functions:

- Manage the Command Interface
- Perform an Automated Predeployment Checkout of the Deployer Systems
- Execute the Mission Profile to Deploy and Retrieve the Satellite
- Manage Telemetry Data
- Support Post Retrieval Checkout of the Deployer
- Manage Ground Operations

Deployer Flight Software

Deployer Flight Software Context Diagram



Deployer Flight Software

Flight Software Components

The Deployer Flight Software Consist of 3 Main Computer Program Components (CPCs), the Operating System CPC, the Flight Processor CPC, and the Event Processor CPC.

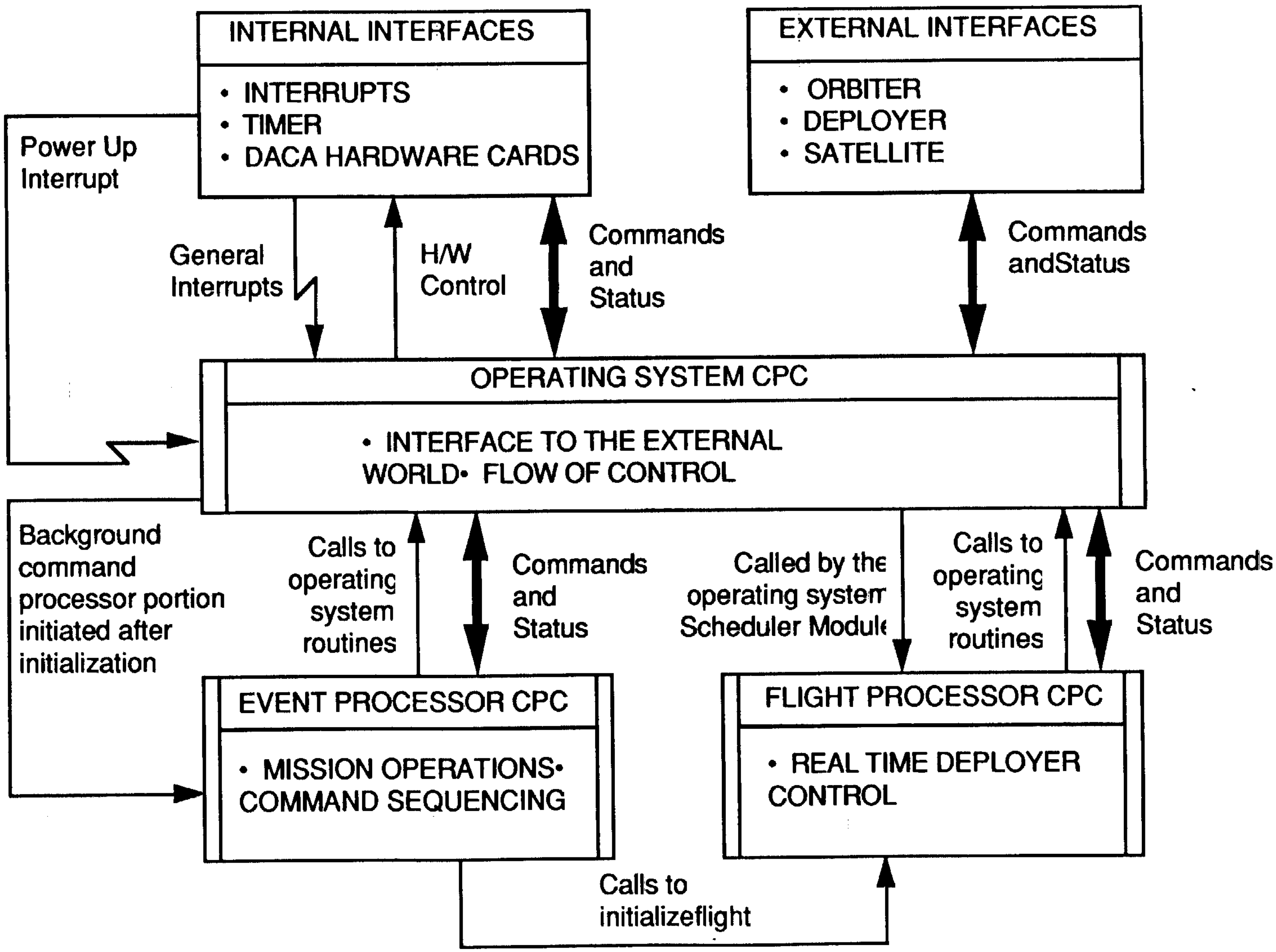
The Operating System CPC Interfaces the Deployer Flight Software to the DACA Hardware and Coordinates the Flight Software Control Flow.

The Flight Processor CPC Executes the Mission Profile Through Closed Loop Feedback Control.

The Event Processor CPC Contains a Command Processor that Interprets and Executes Command Sequences Based on Command Inputs. An Automatic Processor Executes Command Sequences Based on Monitored Events.

Deployer Flight Software

Deployer Flight Software Flow Diagram



Deployer Flight Software

Deployer Flight Software Operational Priority and Sizing & Timing Margins

The Deployer Flight Software Function with a Fixed Priority System which Places the Highest Priority on Interrupt Services, Then Fixed Time Routines, and Then Background Tasks.

The Software Sizing is Summarized Below:

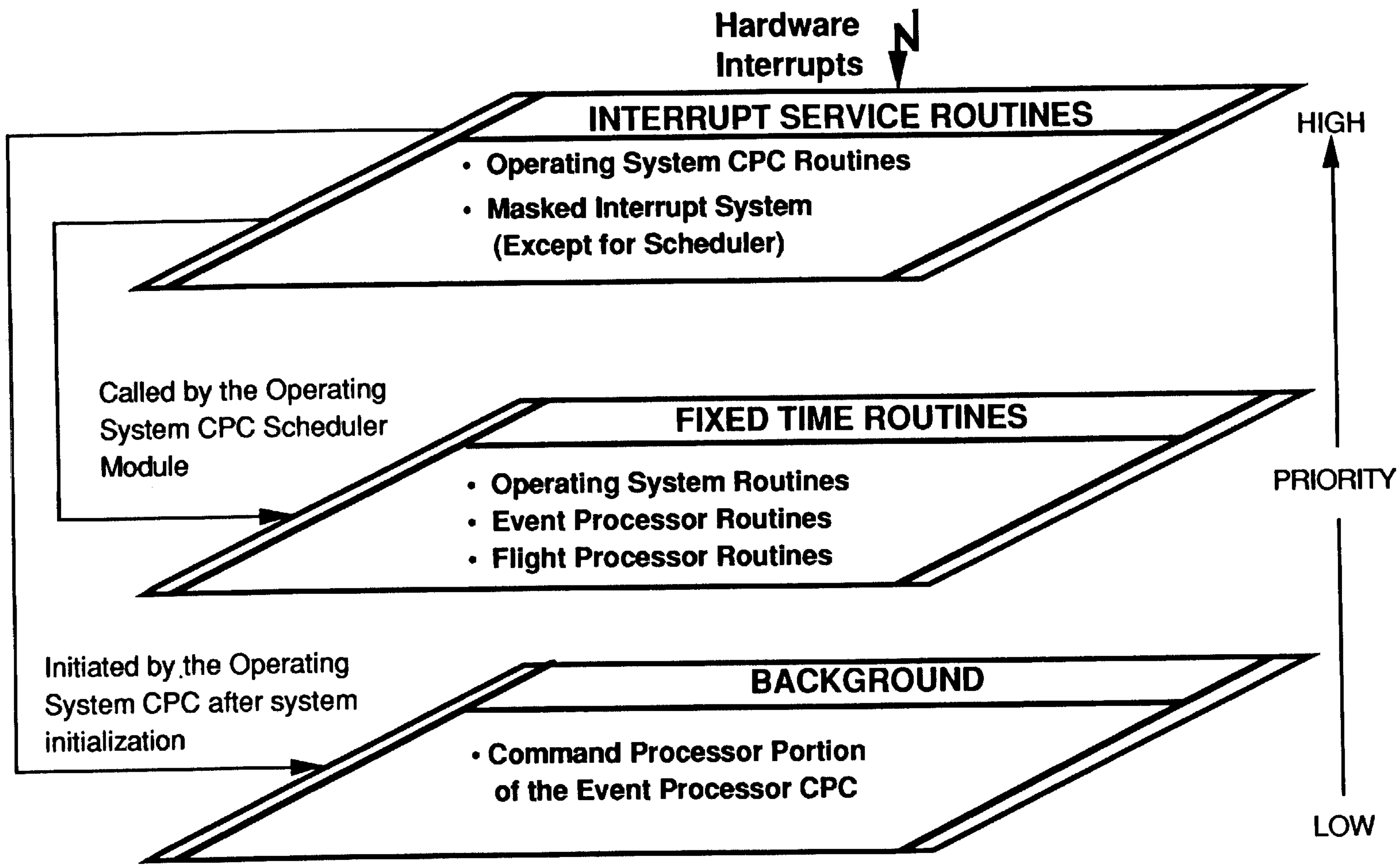
- 24 Modules Consisting of 123 Routines
- 4540 Lines of Executable Code
- 128k Available PROM, 69k Used
- 16k Available RAM, 7.3k Used

The DACA Computational Cycle is 100 mSec Long

Fixed Time Routines - Control Laws, Status Monitoring, etc.	50 mSec
Other Tasks - SFMDM TLM Processing, Satellite TLM, MCA Command & Status, etc.	15 mSec
Total Used	65 mSec
Available Margin	35 mSec
Margin Requirement	20 mSec

Deployer Flight Software

Deployer Flight Software Operational Priority



Pyrotechnic System

Description

The Deployer Pyrotechnic System Activates the Tether Cutting and Boom Ejection Function During Contingency Operations When Conditions Require.

The Pyrotechnic Functions are Controlled by the Crew via Redundant Deployment Pointing Panels (DPPs) in the Aft Flight Deck.

The Primary and Secondary Portions of the System Use Redundant Power Sources, Connectors, Wiring, Pyrotechnic Initiator Controllers (PICs), and NASA Standard Initiators (NSIs).

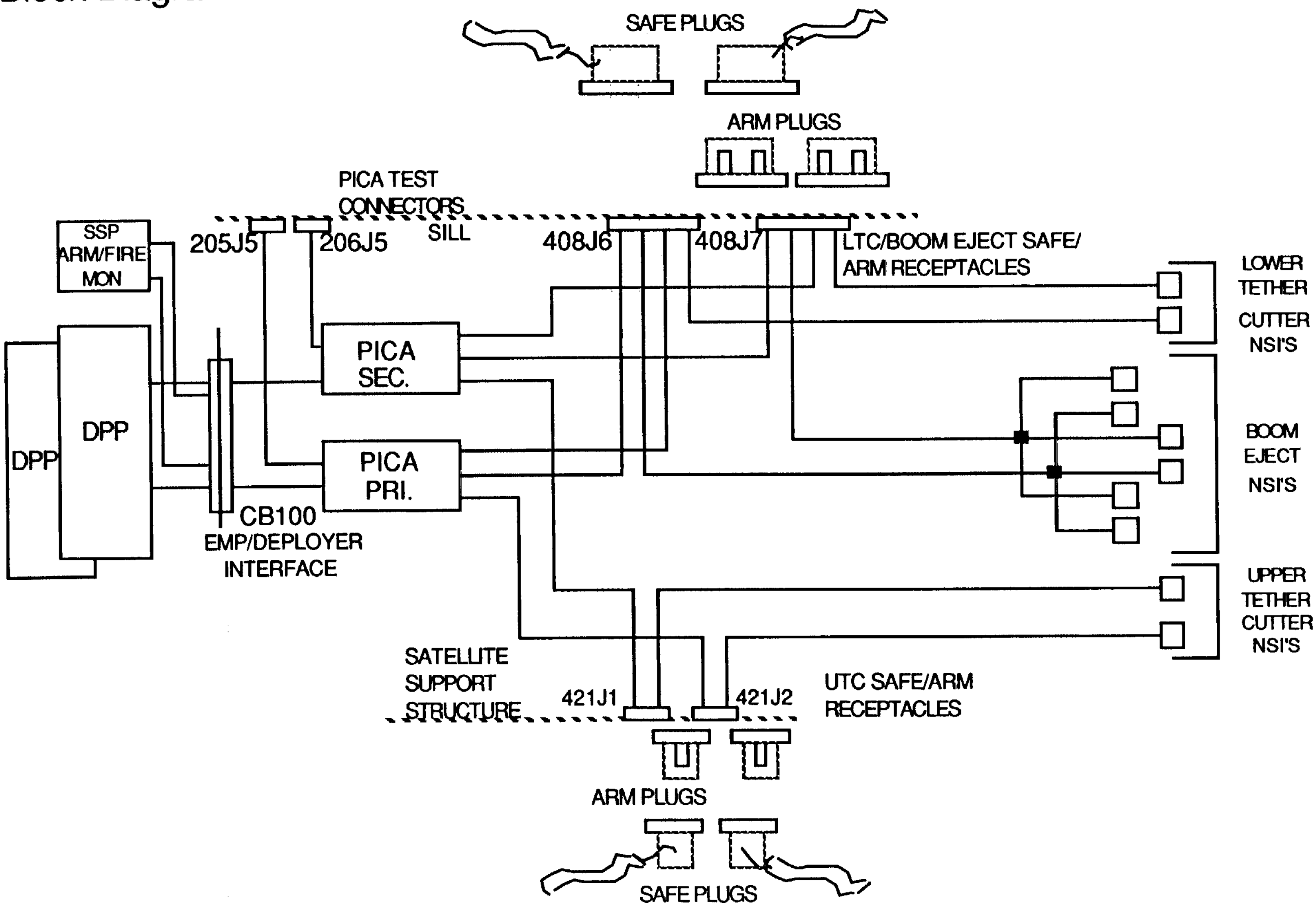
A Total of 10 PICs and NSIs are Utilized as Follows:

- 2 PICs/NSIs for Lower Tether Cutter
- 2 PICs/NSIs for Upper Tether Cutter
- 6 PICs/NSIs for Boom Separation Nuts (3 Nuts Total)

The System Utilizes Safe/Arm Connectors for Safing and Final Checkout Prior to Flight.

Pyrotechnic System

Block Diagram



Pyrotechnic System

Pyrotechnic Initiator Controller Assemblies

The Deployer Contains 2 Pyrotechnic Initiator Controller Assemblies (PICAs)which are Utilized for the Deployer Contingency Pyrotechnic Operations (Tether Cutting and Boom Ejection)

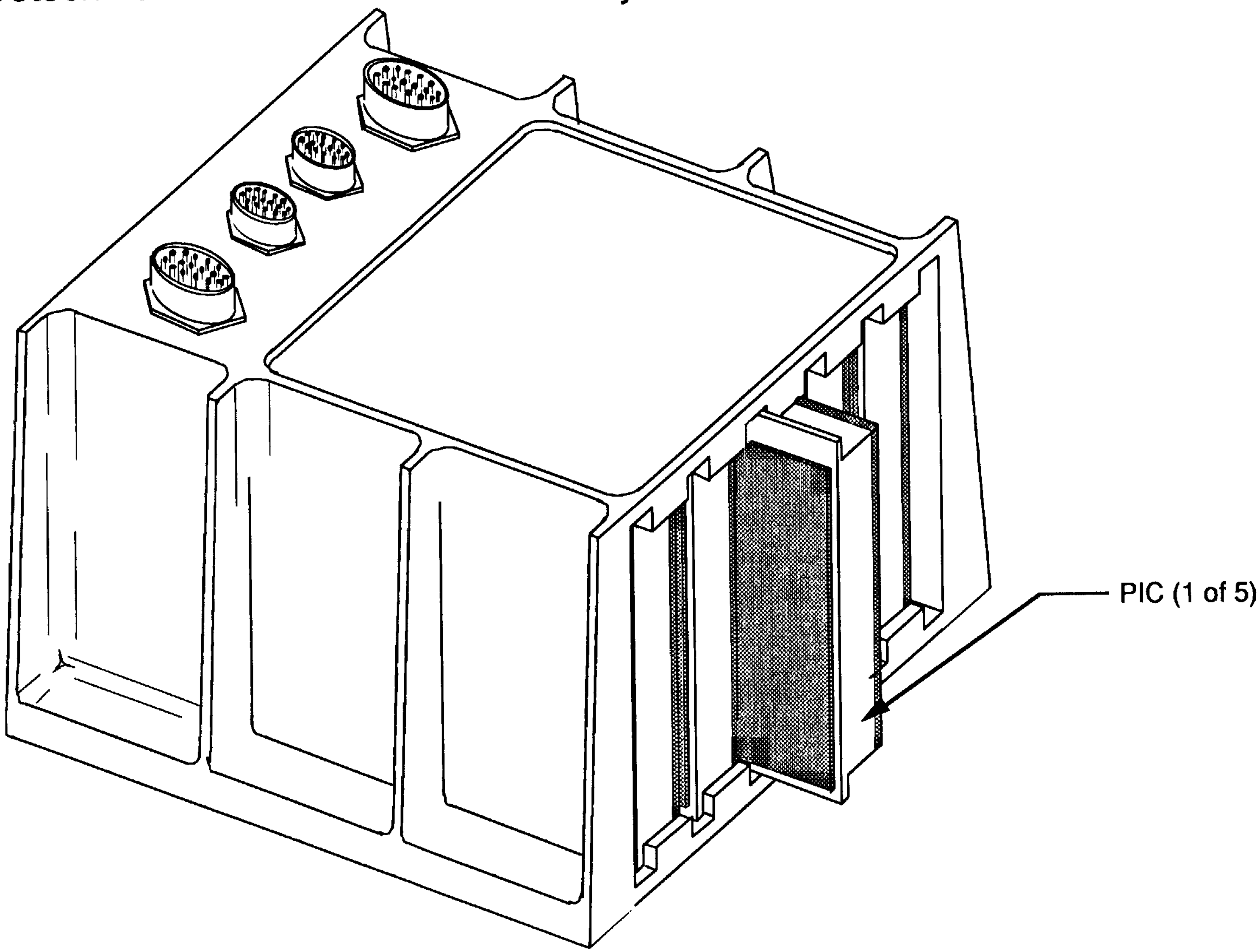
Each PICA Contains 5 Pyrotechnic Initiator Controllers (PICs). Each PIC Delivers Sufficient energy to Provide Sure Fire Detonation of 1 OF THE 2 NASA Standard Initiators (NSI) on Each Device.

The PICAs are Coldplate Mounted for Active Thermal Control.

Dimensions:	7.6 in. x 5.3 in. x 7.6 in.
Weight:	11.8 lb

Pyrotechnic System

Pyrotechnic Initiator Controller Assembly



Deployer Test Program

Component Test Overview

The Deployer Was Tested Both at the Component Level and the System Level.

28 Individual Items Were Tested at the Component Level. Tests Were Either Joint Qualification and Acceptance Tests (JQAT) on Protoflight Units or Separate Qualification and Acceptance Tests on Qual Units and Flight Units. These Items and the Tests that Were Performed on Them are Shown in the Following Table. Performance and Functional Tests Were Performed on All Units.

Deployer Test Program

Component	Random Vibe	Therm-Vac (10^{-5} Torr)	EMI/EMC	Life Tests Run-In Test
Reel Mechanism	JQAT	JQAT	JQAT	N/A
Satellite Restraint Latch	Qual	Qual	System	Qual, Accep
Gearmotor	Qual, Acc	Qual, Acc	Qual	Qual, Accep
U1 & U2 Mechanisms	JQAT	JQAT	System	N/A
Umbilical Connector	Qual	Qual	System	Qual
Boom Assembly	JQAT	EU, JQAT	JQAT	JQAT
Boom Eject Mechanism	N/A	System	System	N/A
Docking Ring	JQAT	JQAT	System	N/A
Tether	N/A	Qual,	N/A	Break Tests
Reel Motor	JQAT	JQAT	System	JQAT
Reel Brake Mechanism	JQAT	JQAT	System	N/A
Brake Motor	Qual, Acc	Qual, Acc	System	Qual, Accep
Hold-Off Brake	Qual, Acc	Qual, Acc	Qual	Qual
LTCM	JQAT	JQAT	System	N/A
Encoder	JQAT	JQAT	System	JQAT
UTCM	JQAT	JQAT	N/A	N/A
Vernier Motor	JQAT	JQAT	JQAT	JQAT
Concentric Damper	JQAT	Qual, JQAT	N/A	Qual
DACA	JQAT	JQAT	JQAT	JQAT
MCA	JQAT	JQAT	JQAT	JQAT
MPC	JQAT	JQAT	JQAT	JQAT
HVRA	System	System	System	N/A
PICA	Qual, Acc	Qual, Acc	System	N/A
Tether Cutter	Qual, Acc	Qual, Acc	System	N/A
Boom Separation Nuts	Qual, Acc	Qual, Acc	System	Proof Load

Deployer Test Program

System Level Testing

The Deployer System Level Testing Was Performed at MMAG Following Assembly and at KSC Following Shipment and Integration.

Testing at MMAG Included a Modal Survey, Hardware/Software Integration Tests (HSIT), EMI/EMC, Acoustic, Thermal Balance, and Final Performance.

The System Test Matrix is Included on the Following Pages. The Top of the Matrix Lists the Functional Test, and the Left Side of the Matrix Lists the Major Tests and Sub-Tests. If a Sub-Test Was Performed, an "X" is in the Appropriate Row/Column.

The System Level Tests Were Divided into 7 Major Tests, Each with Sub-Tests. For the HSIT, Functionals, Thermal Balance, and Performance Tests Specific Sub-Tests Were Performed to Verify System Operation. Likewise, the KSC Testing Involved Specific Sub-Tests.

The Following Tests are Shown on this Page of the Matrix:

1. Initial Operations
   Functionally Verify Deployer Operation

2. Brake Operations
   Verify Total Brake System is Operating Per the Requirements

3. Detail Operations
   Verify All Commands and Monitors are Operational and Meet Specifications and That All Scale Factors are Correct. Assure Safety Inhibits are Operational.

Deployer Test Program

System Functional Test Matrix

		MODAL	HSIT EDU DACA	HSIT FLT DACA	EMI	RING MOD.	FUNC	U1/U2	ACCOUS	FUNC	BOOM	PRE T/V	T/V HOT/COLD	BOOM	FINAL PERF.	KSC OFF-LINE	KSC LEVEL IV	KSC PPCU	KSC LEVEL I (CITE)	KSC PAD
1.0 INITIAL OPS.	a. ISOLATION		X	X			X					X				X	X	X		
	b. EGSE C.O.		X				X					X				X	X			
	c. PWR SYS		X		X		X			X		X	X		X	X	X	X	X	X
	d. COMM		X	X	X		X			X			X		X	X	X	X	X	X
	e. TENSION		X				X					X			X	X	X			
	f. BRAKE		X	X	X		X			X			X		X	X	X	X	X	
	g. SPOOL		X		X															
	h. SPOOL w/VM		X		X															
	i. MIN/MAX BUS		X																	
2.0 BRAKE	a. MANUAL		X												X		X			
	b. RATES																			
	c. DISABLE		X												X					
	d. LENGTH		A				X								X		X			
	e. SFT																			
	f. PWR OFF		X												X		X			
	g. TENSION		X												X		X			
3.0 DETAIL OPS.	a. SENSOR			X								X			X					
	b. SAT I/F		X	X	X		X			X			X		X	X	X	X		
	c. LAUNCH LOCK		X				X					X			X	X	X			
	d. LATCHES		X	X	X		X			X		X	X		X	X	X	X		
	e. SAT ROT							X		X			X		X	X	X	X		
	f. TETHER		X									X					X			
	g. THERMAL SYS			X			X			X			X		X	X	X	X	X	
	h. CMD/MON																			

LEGEND: X - REQUIREMENT TESTED
A - MAJOR ANOMALY

Deployer Test Program

System Tests - Overview/Purpose

The Following Tests are Shown on this Page of the Matrix:

4. Design Reference Mission
   Verify System Control of Satellite by Using Motor, Feedback, and EGSE.

5. Pyro Operations
   Verify Pyro Functions are Operating Per Requirements.

6. Umbilical Test
   Verify Umbilical Operations Using Satellite Structural Test Model (STM).

7. Boom Operations
   Verify Boom Extension, Retraction, and Associated Functions.

Deployer Test Program

System Functional Test Matrix

		MODAL	HSIT EDU DACA	HSIT FLT DACA	EMI	RING MOD.	FUNC	U1/U2	ACCOUS	FUNC	BOOM	PRE T/V	T/V HOT/COLD	BOOM	FINAL PERF.	KSC OFF-LINE	KSC LEVEL IV	KSC PPCU	KSC LEVEL I (CITE)	KSC PAD
4.0 DRM	a. L.T. FLY-A -WAY		X				X					X	X		X	X	X	X	B	
	b. DEPLOY		X	A													X			
	c. YO-YO		X	X													X			
	d. RETRIEVE		X	X													X			
	e. L.T. DOCK		X				X					X	X		X	X	X	X		
	f. SS /RESUME		X				X								X		L			
	g. UPLINK																			
5.0 PYRO	a. ENERGY			X			X			X		X	X		X	X	X	X	X	X
	b. NO FIRE																			
	c. ARM/SAFE			X			X								X		X		X	X
	d. INHIBITS			X			X								X		X			
	e. PWR/UNPWR			X			X								X		X			
6.0 U1/U2	a. MATE							X							X		X	X		
	b. BATT ACCESS							X									X			
	c. OPERATION							X		X					X		X	X		
7.0 BOOM	a. EXTENSION										X			X						
	b. RETRACTION										X			X						
	c. INTER. STOP										X			X						
	d. SFT										X			X						

LEGEND:
- X - REQUIREMENT TESTED
- A - MAJOR ANOMALY
- B - RM & VM CHECK OUT ONLY
- L - LOAD FLIGHT TETHER

Deployer Test Program

2.0 Brake Ops Nominal Test Results

The Brake Operations Tests Verified that the Brake System is Operating Per the Requirements.

Tests Included High Rate, Low Rate, and Length Brake Trips with the Tether Travelling in Both Directions. These Tests Were Performed at the Beginning of the System Level Tests and Again at the End.

Conclusion: Brake Trips Were Nominal with No Change During System Level Testing

Deployer Test Program

2.0 Brake Ops Nominal Test Results

Test	Limits	Pre Env. Circuit 1	Pre Env. Circuit 2	Post Env. Circuit 1	Post Env. Circuit 2
Deploy Direction					
Low Rate (m/s)	0.09 - 0.105	0.096	0.097	0.098	0.096
High Rate (m/s)	0.37 - 0.43	0.39	0.39	0.39	0.38
Length (m)	220 max	164	164	164	164
Retrieve Direction					
Low Rate (m/s)	0.09 - 0.105	0.099	0.098	0.098	0.097
High Rate (m/s)	0.37 - 0.43	0.39	0.38	0.38	0.38
Length (m)	220 max	184	184	183	183

Deployer Test Program

3.0 Detail Ops Nominal Test Results

The Detail Ops Tests Verified that all the Commands and Monitors Were Operational and that the Scale Factors Were Correct.

Tests Were Included to Measure Tether Tension and Length Error, Both Prior to and After Environmental Exposure. The Satellite Restraint Latch Preload Was Also Checked Both Prior to and After Environmental Exposure.

Conclusion: No Detectable Change in Tension Measurements, Latch, Launch Lock, Satellite Rotation or Temperature Measurements.

Deployer Test Program

3.0 Detail Ops Nominal Test Results

Test	Limits	Pre Env.	Post Env
Tension Error			
LTCM	±4 %	7.3 %*	8.2 %*
UTCM - C	±4 %	3.3 %	2.2 %
UTCM - F	±13 %	4.4 %	4.4 %
Length Error	±5 %	0.231	N/A
Latch Preload	3400 min	3900 - 4190	3760 - 4010
Total Operations - 27 min			

* Includes Error Due to UTCM Internal Friction

Deployer Test Program

4.0 Design Reference Mission Test Results

The Design Reference Mission Tests Included 24 Fly-aways and Docks, and 2 Complete Mission Simulations Using the Deployer Tether Control System and the Ground Support Equipment. A Total of 314,000 Meters of Tether Was Moved.

Tests Were Included to Measure System Friction and Margins.

Conclusion: System Margins are Adequate, No Detectable Changes Over Environments, No Inadvertent Brake Trips.

Deployer Test Program

4.0 Design Reference Mission Test Results

Test	Total
Fly-away & Docks	24
Full Missions	2
Contingency Missions	3
Total Tether Moved	314,000 m
Fly-away Friction Margin	
Pre Environmental	85.8 %
Post Environmental	101.3 %

Deployer Test Program

5.0 Pyro Ops Test Results

The Pyro Operations Test Verified Proper Operation of the Pyrotechnic System, Including Resistance Checks and Energy Levels.

The Test Configuration Used is Shown.

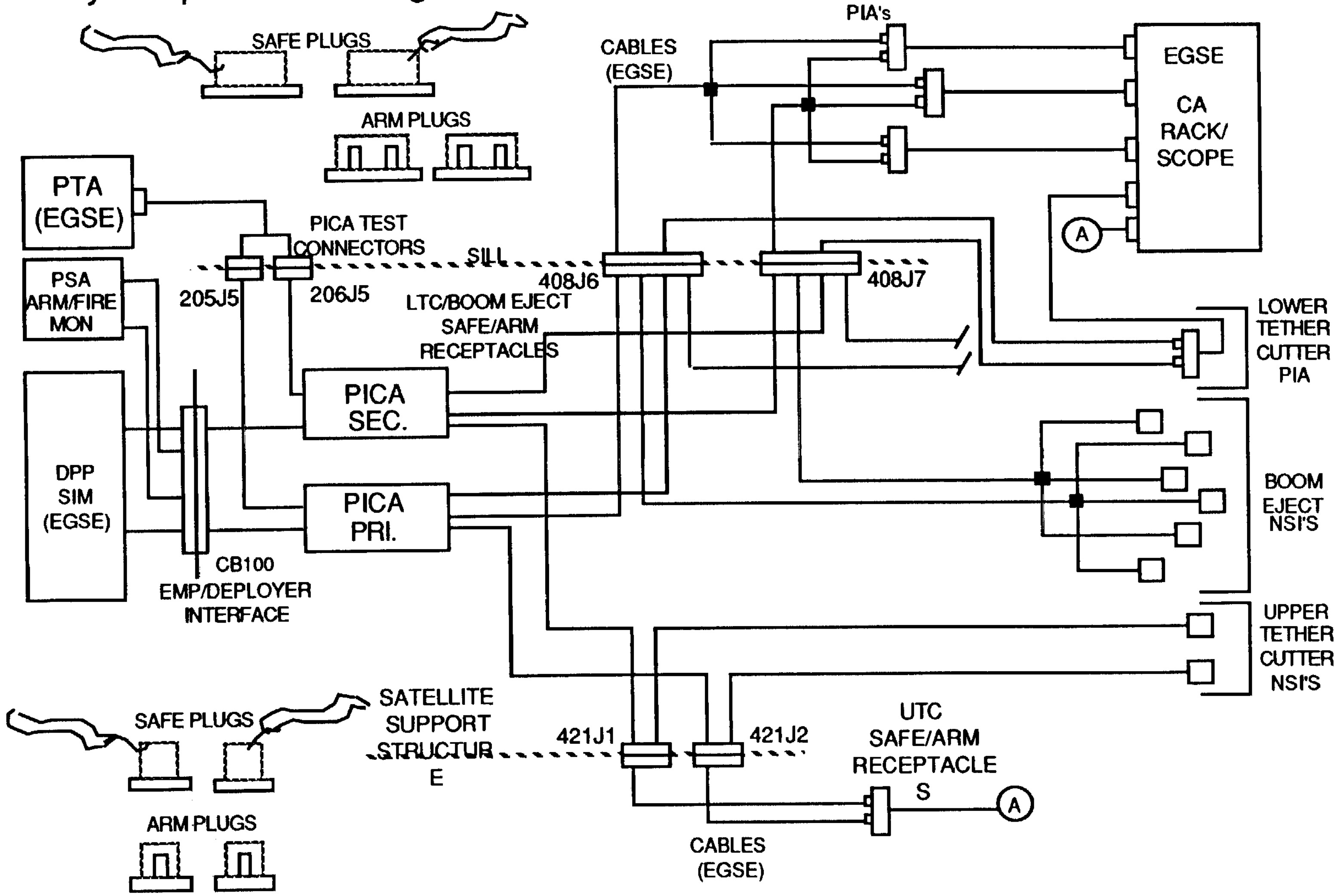
Conclusion: All Pyros Fired at the Proper Time.
Resistance Test Passed.
Pyro Energy Exceeds Requirements.

The Umbilical Operations Test Verifed that the Umbilicals Can be Mated to the Satellite and that the Umbilical Mechanisms can Disconnect and Retract the Umbilicals. These Tests Were Performed with the Satellite Structural Test Model (STM). Access for Installing the Satellite Batteries Was Also Verified.

Conclusion: Umbilicals Can Be Mated, Both Pre & Post Environmental.
Umbilical Retraction and Stow Pre & Post Environmental.Was Successful.
Access for Satellite Battery Installation is Sufficient.

Deployer Test Program

5.0 Pyro Ops Test Configuration



Deployer Test Program

6.0 Umbilical Ops Nominal Test Results

The Umbilical Operations Test Verifed that the Umbilicals Can be Mated to the Satellite and that the Umbilical Mechanisms can Disconnect and Retract the Umbilicals. These Tests Were Performed with the Satellite Structural Test Model (STM). Access for Installing the Satellite Batteries Was Also Verified.

Conclusion:

Umbilicals Can Be Mated, Both Pre & Post Environmental.
Umbilical Retraction and Stow Pre & Post Environmental.Was Successful.
Access for Satellite Battery Installation is Sufficient.

Deployer Test Program

6.0 Umbilical Ops Test Results

Test	Results	T/V Cold Drive Time	T/V Hot Drive Time
Umbilical Mate	Successful		
Umbilical Retract	Successful		
Satellite Rotation		00:03:34	00:02:41

Deployer Test Program

7.0 Boom Ops Nominal Test Results

The Boom Operation Tests Included 3 Complete Extension/Retraction Cycles After Acoustic Vibration and 3 Complete Extension/Retraction Cycles After Thermal Vac. A Total of 8 Partial Extension/Retraction Cycles Were Also Performed.

Tests Were Included to Verify Proper Functioning of the Intermediate Boom Stop.

Conclusion: Boom Extends and Retracts at the Proper Rates Before and After Exposure to Flight Environment Conditions. System Redundancy and Intermediate Stop Operate Properly.

Deployer Test Program

7.0 Boom Ops Nominal Test Results

	Limits	Pre Env.	Post Env.
Motor Current (Amps)	5.0	2.8	2.7
Rate (m/s)			
Extend	0.02	0.015	0.015
Retract	0.02	0.015	0.015
Brake Tension (N)	40 min	75	75

Deployer Test Program

Thermal Balance Overview

The Deployer Was Subjected to Thermal Vacuum Conditions for 231 Hours.
- 10^{-6} Torr
- 1.1 Suns Max
- -190°C Chamber Wall Temp Min

The Thermal Balance Test (TBT) Included the Following "Hot" and "Cold" Functional Tests:
- Launch Lock Operation
- Latch Operation
- Docking Ring Rotation
- Thermal System Monitoring
- Low Tension Flyaway
- Low Tension Docking
- Pyro Energy

The Planned Structural Temperature Range for the Deployer Reel Bearings of -40°C to +40°C Was Achieved, with a Dwell Time of 8 Hours at Each Extreme.

The Thermal Balance Test Verified Deployer Heater Operation and Coolant Loop Design and Provided Data to Update the Deployer Thermal Math Model.

Deployer Test Program

Thermal Balance Timeline

Test No.	Test Title	Duration
1	Launch Initialization & System Checkout	12 hrs (Approx)
2	Transient Cool Down to On-Orbit Quiescent	48 hrs
2a	Hot Nest Operation	8 hrs
3	Solar Inertial Transient	1 hr
3a	Recovery From Solar Inertial	6 hrs
3b	Hot Nest Off Transient	6 hrs
3c	Transition to Max Cold	8 hrs
3d	Load Bank Test	4 hrs (Approx)
4	Deep Space Viewing	1.5 hrs
4a	Deep Space Recovery	8 hrs
5	Cold Functionals	8 hrs (Approx)
6	Transient to -30°C Chamber Wall	12 hrs (Approx)
6a	MLI Steady State Effectiveness Test	48 hrs (Approx)
7	Transition to Max Hot	12 hrs (Approx)
7a	Load Bank Test Max Hot	4 hrs
8	Hot Functionals	8 hrs (Approx)
8a	Brake Motor Hot Operation	4 hrs

Deployer Test Program

Modal Survey

The Deployer Modal Survey Was Completed in January 1990. All Resonant Frequencies Were Within 5% of the Test Frequencies for all Target Modes. The Cross-Orthogonality Was Greater Than 0.9 for All Target Modes.

Changes Were Made to the Deployer Mathematical Model as a Result of the Modal Survey. These Changes Include:

1) Updated all mass Properties to Match Actual Weight.
2) Reduced the Modulus of Elasticity of Struts, 426 Ring, and Reel Shaft by 10% and Upper SSS Skin by 5%.
3) Included Detailed Models of Strut End Fittings.

Deployer Test Program

EMI/EMC

The Deployer EMC Testing Was Completed in April 1990. The Purpose of the Test Was to Verify Compliance with the Deployer Specification, TSS-CEI-02. The Test Was Performed in Accordance with MIL-STD-461A and MIL-STD-462, Tailored to NASA ICD 2-19001.

The Deployer Demonstrated Compliance with the Qualification Criteria for CE01, RE01, RS03, Ripple, CS02, CS06, RS01, Radiated Susceptibility BB and RS03.

The Deployer Was Non-Compliant with CE03. The Majority of the Emissions Were Attributed to the EGSE. The Remaining Emissions Exceeded the Limit by 3 dB Between 4 to 7 MHz. Sufficient Margin Exists to Preclude any Intrasystem EMC Problems Due to Conducted EMI.

Radiated Ambient Emissions Exceeded the RE02 Specification Limits. The Detected Emissions Were Determined to be from the EGSE. Emissions Attributable to the Deployer are not Within the Orbiter Critical Telemetry or Communication Frequencies. The Margin Between these Emissons and the Radiated Susceptability Requirement is Greater Than 130 dB Which is Sufficient to Preclude any System EMC Problems Due to Radiated EMI.

Deployer Test Program

Acoustic Test

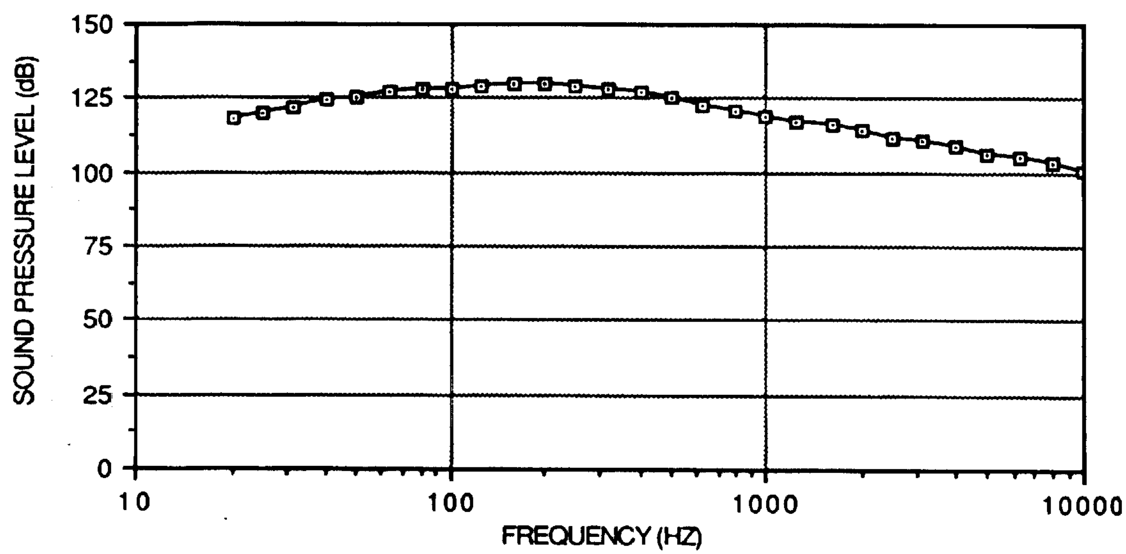
The Deployer Acoustic Test Was Performed in June 1990 in the Reverberant Acoustic Laboratory. The Test Objectives Were to Show Compliance with the Deployer Acoustic Noise Requirements in TSS-CEI-02, to Verify Structural Integrity and Workmanship, to Demonstrate That There is No Degradation in System Performance Following Exposure, and to Obtain Data for evaluating TSS Zonal Vibration Criteria.

The Deployer Was Exposed to the Spectrum Shown, Which Was Taken from the SPAH. It Represents the Combined Launch and Aeronoise Spectra. The Overall Sound Pressure Level Was 139 dB, and the Deployer Was Subjected to this Level for 60 Seconds.

All Test Objectives Were Met. The Deployer Incurred no Structural Damage During the Test. The Only Anomaly Was Several Screws on the Backshell of a Connector Backed Out. The Screws Should Have Been Bonded in Place Prior to Testing, But Were Not. These Screws Were Properly Installed After the Test. All Mechanisms and Electronics Worked Properly, and No Degradation in Their Performance Was Noted in the Post Acoustic Functional Testing.

Deployer Test Program

Acoustic Test Spectrum



Electrical Ground Support Equipment

EGSE

The EGSE Consists of 3 Racks and a Micro-Computer that Provide Test Simulation of the Deployer Interfaces and Satellite Dynamics.

The Controller Assembly (CA) Consists of the Following:

- 8086 CPU, an 8087 Coprocessor, 128 kBytes of ROM, and 512 kBytes of RAM Which Store and Execute the Simulation of the Deployeed Satellite Dynamics.
- An Optical Disc Provides 400 MBytes of Data Storage, Which Translates to 70 Hours of DACA Run Time.
- The Time Code Generator Which Simulates the Orbiter GMT.
- The SFMDM Simulates the SFMDM SDIO Channel to the DACA.
- The Satellite Interface the Discrete Signals To/From the Satellite, the Telemetry to the DACA, and the Serial Commands from the DACA.

The Power Switching Assembly (PSA) Provides Simulation of the Following Orbiter Power and Discrete Commands:

- Manual Switches for the SSP Functions.
- A Single DPP for Pyro Functions.
- Orbiter PF-1 MDM Simulation by Manual Switches.
- SSP and PF-1 MDM Monitors by Panel Lights.

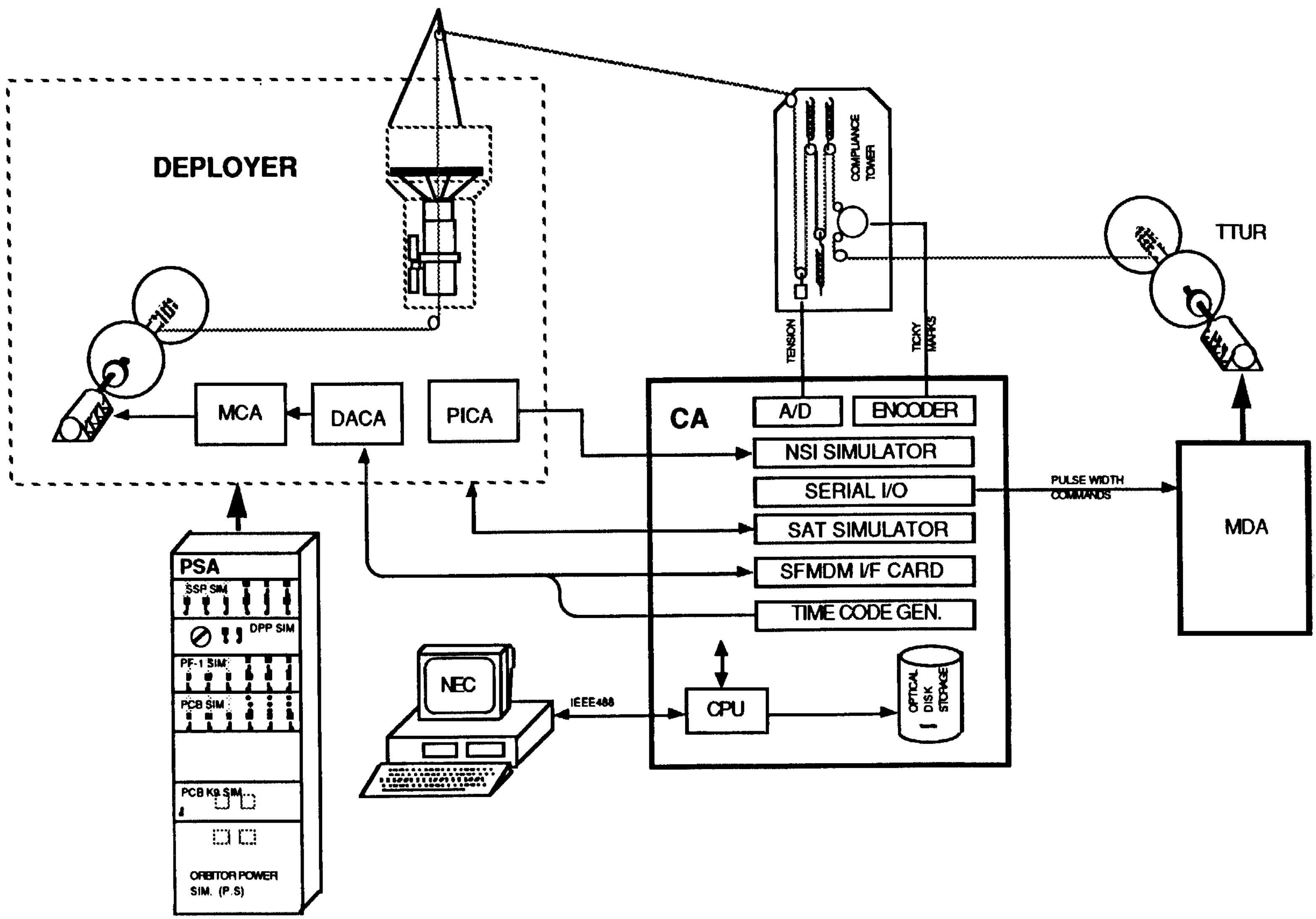
The NEC Microcomputer Provides the Following:

- Operator Interface to Control the Satellite Dynamics Simulations and the Deployer
- Post Test Data Processing and Real Time Test Data Display
- CA Software Loading

The Motor Drive Assembly (MDA) Provides Pulse Width Modulated Power to Drive the Tether Take Up Reel Motor. It is Driven by the CA Rack.

Electrical Ground Support Equipment

EGSE



4. Orbital Dynamics

Orbital Dynamics

Introduction

The Major Objective of TSS-1 is to Validate Operation of the Reusable Tethered Satellite System Including Closed-Loop Control of the 20 km (12 mile) Deployment and Retrieval of the Satellite.

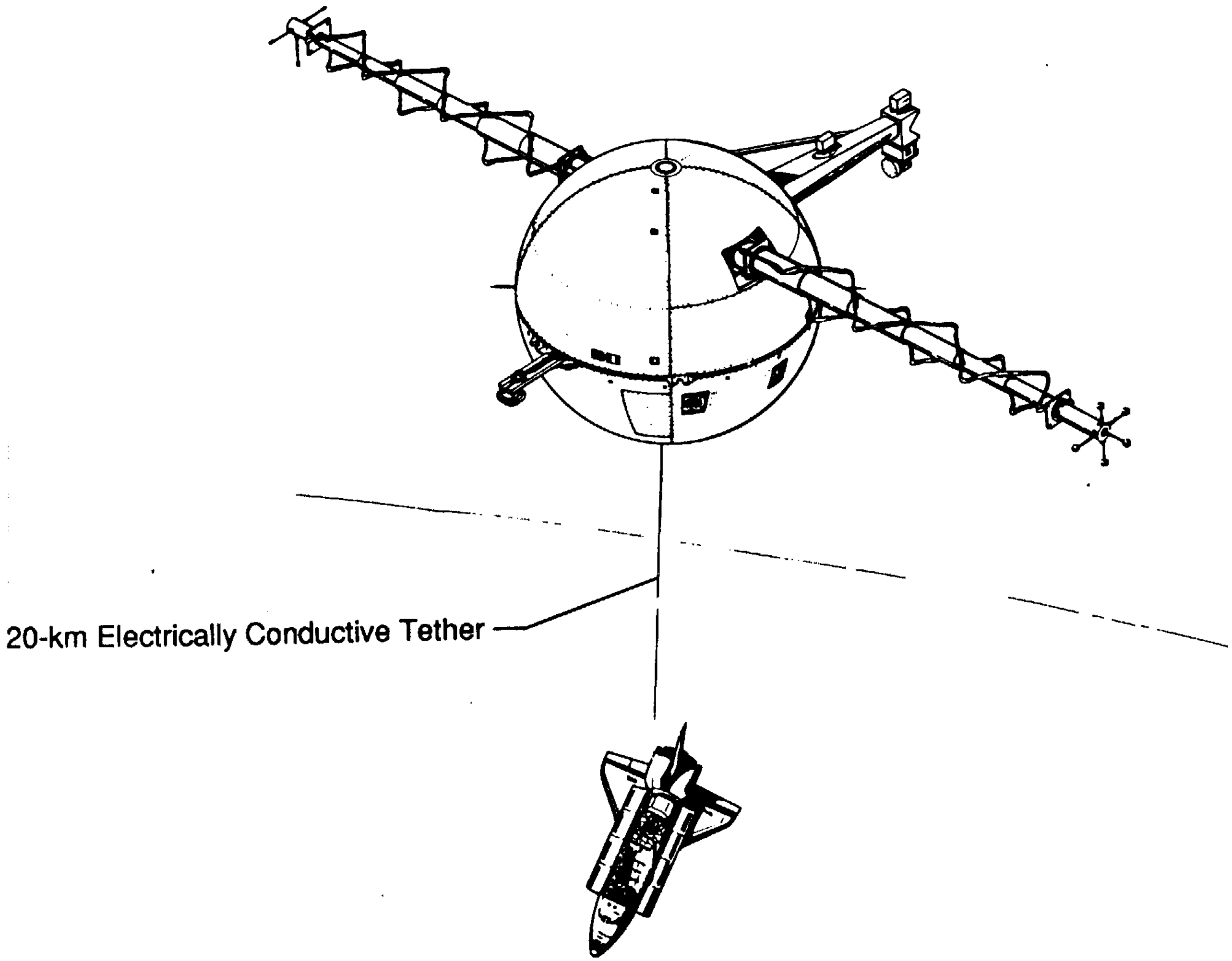
Scientific Instruments are Used in the Orbiter Bay and on the Satellite in Support of the Electrodynamics Investigation. This Involves Electrical Power Generation as the Conductive Tether Intersects the Earth's Magnetic Field.

Gravity Gradient Forces and Torques Provide for a Stable Configuration with the Satellite Deployed Up or Down. The Forces Generated by the Gravity Gradient are Small. For the TSS-1 Mission, the Force is ≈ 50 N When the Satellite is On-Station at 20 km.

Forces are Also Induced by the Electric Current Flowing in the Tether. These Forces Induce a Skip Rope Like Motion in the Tether.

Orbital Dynamics

Introduction



Orbital Dynamics
Overview

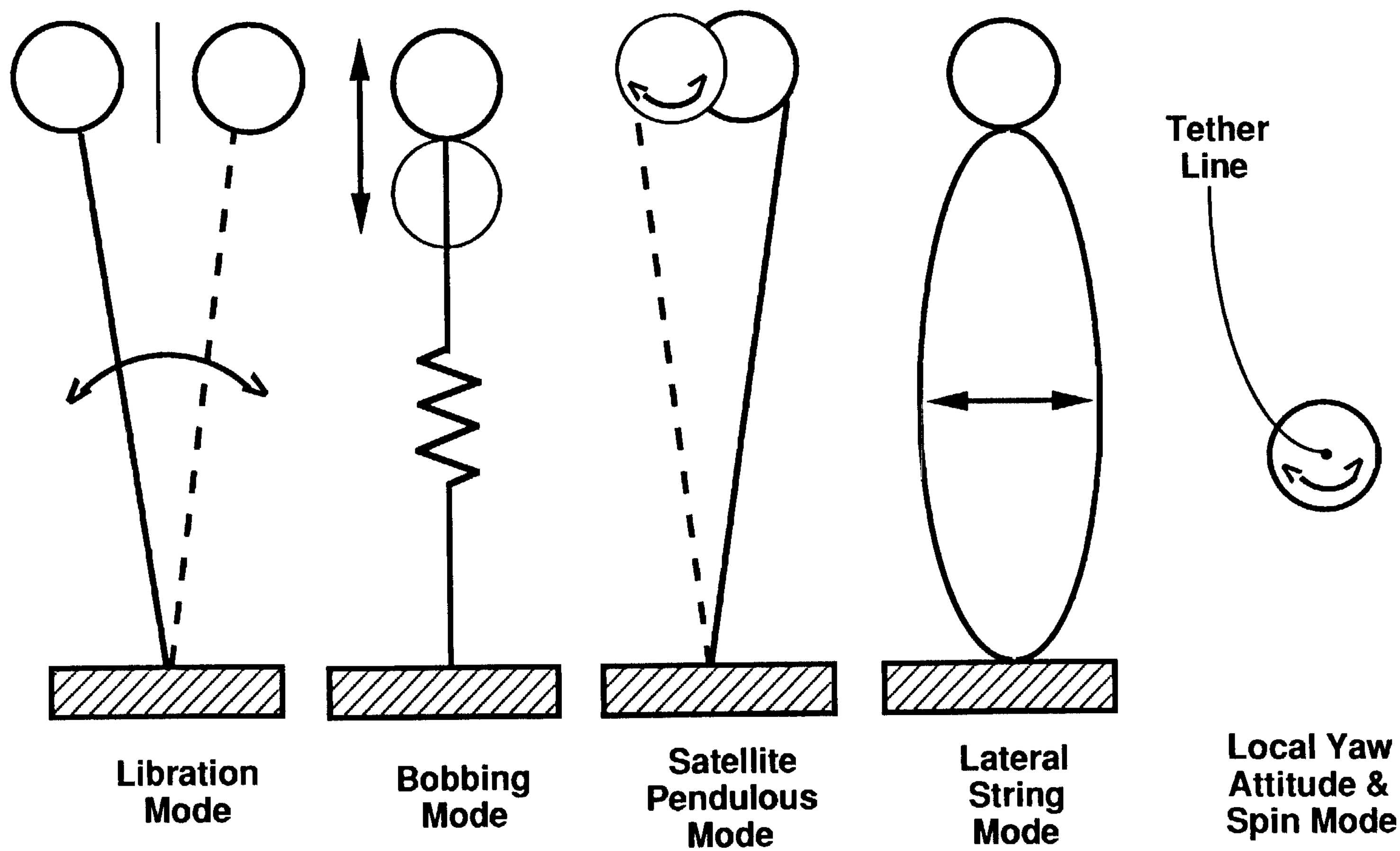
The Satellite Can Lose Stability if Deployment or Retrieval Velocities are too High. The System Swing Angle in the Orbital Plane is a Function of the Tether Rate. An Excessive Tether Rate Can Cause this Angle to Approach 90°, Which May Cause the System to Go Unstable.

The TSS-1 Satellite has Automatic Attitude Control about the Yaw Axis Only. This Allows for Possible Attitude Dynamics Problems, Particularily String Dynamics Interaction with Satellite Dynamics at a Tether Length of 400 meters.

Many of the Frequencies Associated with Tethered Systems in Space are Very Low. In Low Earth Orbit, the Orbit Period is ≈90 Minutes. The System Swing Angle in the Orbital Plane is Called In-Plane Libration and Has a Period of 53 Minutes. The Out-of-Plane Swing Angle is Called Out-of-Plane Libration and Has a Period of 45 Minutes. The Period of the First Skiprope Mode is 3 Minutes at a Distance of 2.4 km.

Orbital Dynamics

Overview



Orbital Dynamics

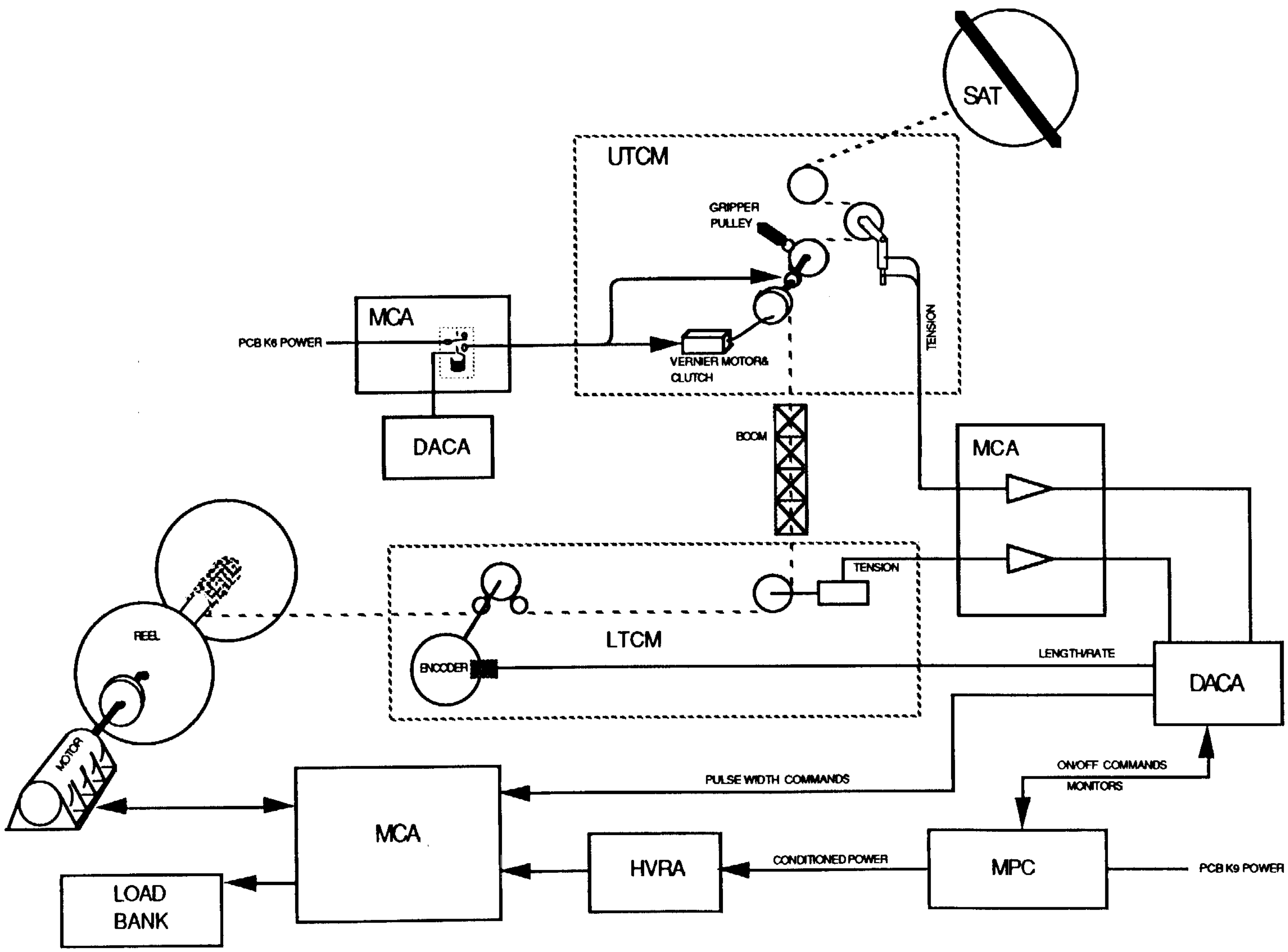
Deployer Control Characteristics/Description

The Deployer Control System is Required to Reel the Tether In or Out in Accordance with the Established Mission Profile. The Mission Profile is Capable of Controlling the In-Plane Libration and the Extensional Mode, or "Bobbing" by Tension Feedback Control or Back EMF Damping from the Reel Motor.

Sensors Include the 2 Stage UTCM Tensiometer, the LTCM Tensiometer, and the Encoder. Effectors (Actuators) Include the Reel Motor, Which Can Operate Both as a Generator and a Motor, and the Vernier Motor which Overcomes Deployer Mechanism Frictions During Deployment.

Command & Data Management

Tether Control System



Orbital Dynamics

Nominal Mission Profile Basis

The Nominal Mission Profile is Based on the Following Assumptions:

The Out-of-Plane Motion Will Remain Null if it is Initialized as Zero.

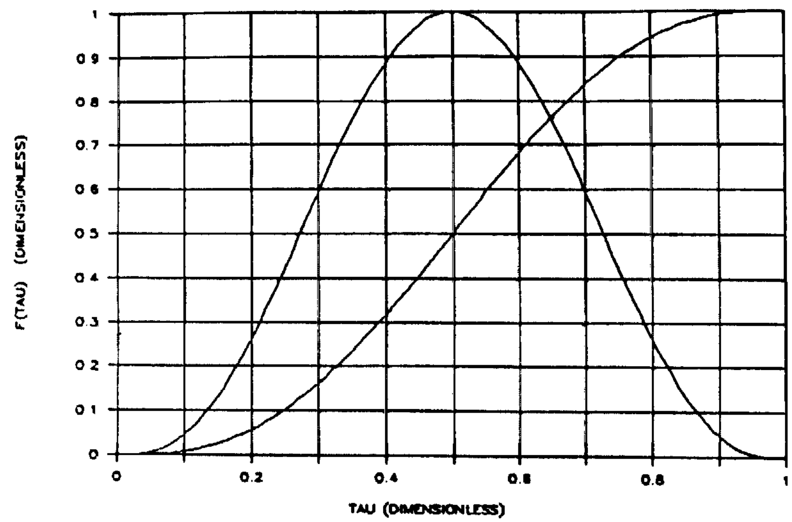
If the In-Plane Libration Angle (q), as Well as its Time Derivatives, is Specified as an Explicit Function of Time, Then an Auxiliary Differential Equation Involving Length and Length Rate Can be Derived. Thus Length and its Time Derivatives Can be Calculated as Explicit Functions of Time.

Given that the Out-of-Plane Angle (f) and its Time Derivatives are Null, and That q , Length, and Their Time Derivatives are Known Explicit Functions of Time, Then the Tension in the Tether, f, Can be Expressed as aTime Dependent Function.

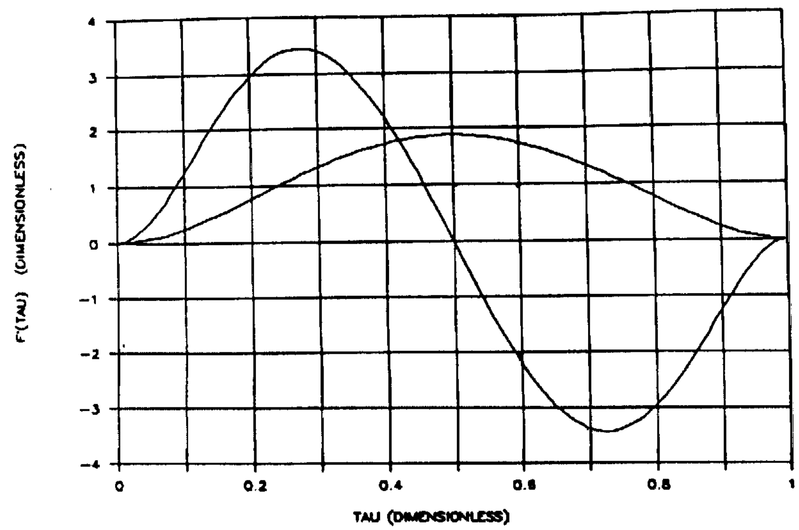
By Implementing the Above Assumptions, Length can be Expressed as a Solvable Differential Function of Time, and Tension can be Expressed as a Dimensionless Function of Time Which Allows the Calculation of a Continuous, or "Moving", In-Plane Angle for the Mission Profile.

These Functions Produce the Synthesized Shape Functions of Commanded q Shown.

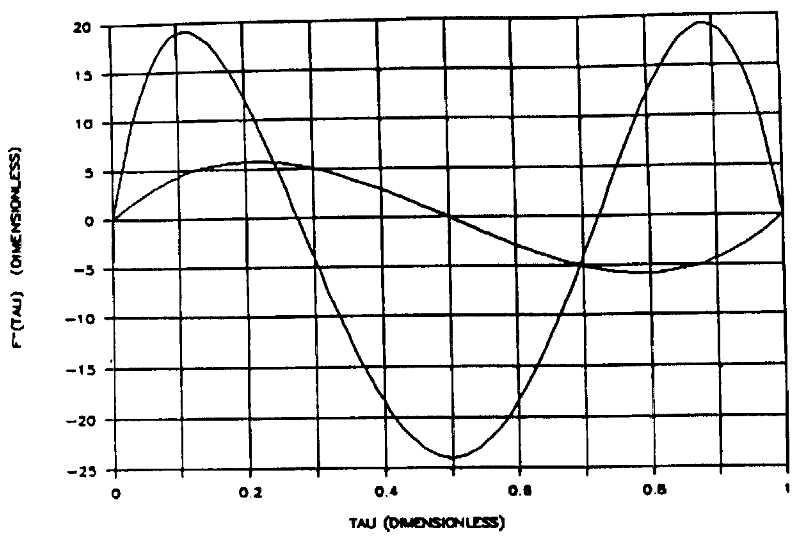
Mission Profile Shape Functions



F(τ) vs. τ for KEY = 2 and KEY = 3



F'(τ) vs. τ for KEY = 2 and KEY = 3



F''(τ) vs. τ for KEY = 2 and KEY = 3

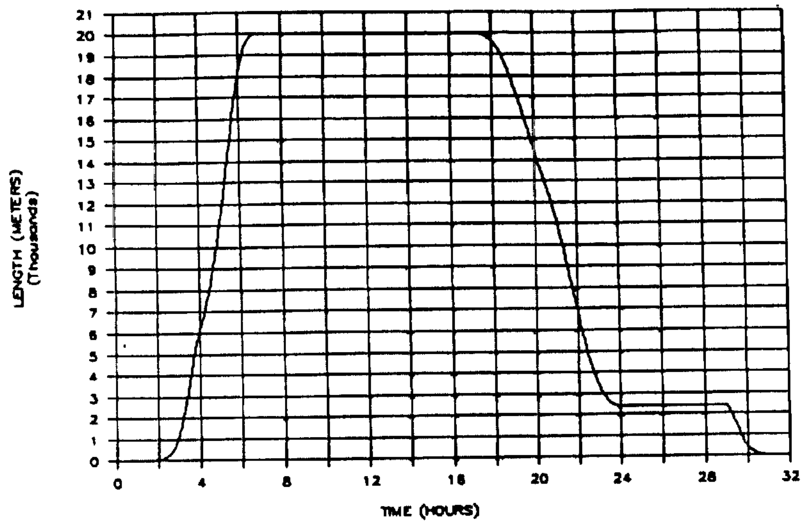
Orbital Dynamics

Nominal Mission Profile

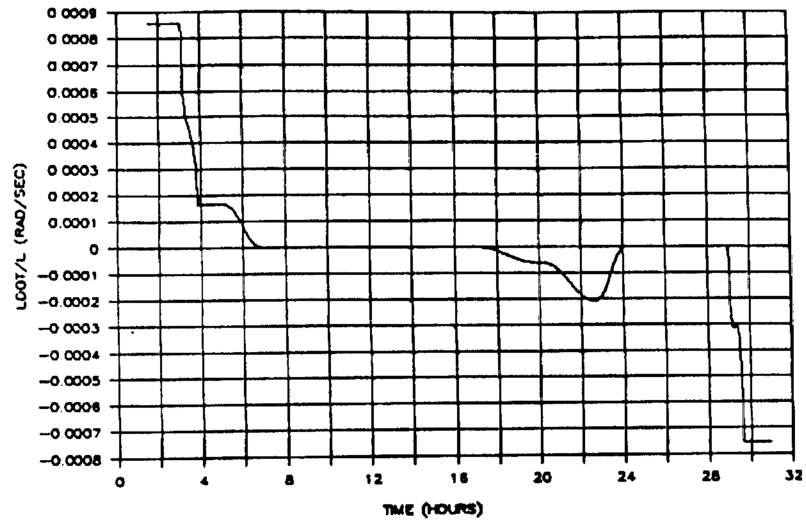
The Mission Profile has Been Developed to Provide for Both Engineering and Science Objectives. The Flight Control/Profiling Algorithm has a ROM Database for Nominal Flight Operations. The Profile Algorithm Can Accept a Ground Uplink Database for Contingency/Alternate Flight Operations. This is Accomplished by Way of a Soft Stop and Resume Capability. The Control Software Provides for Crew Interaction with the System.

Orbital Dynamics

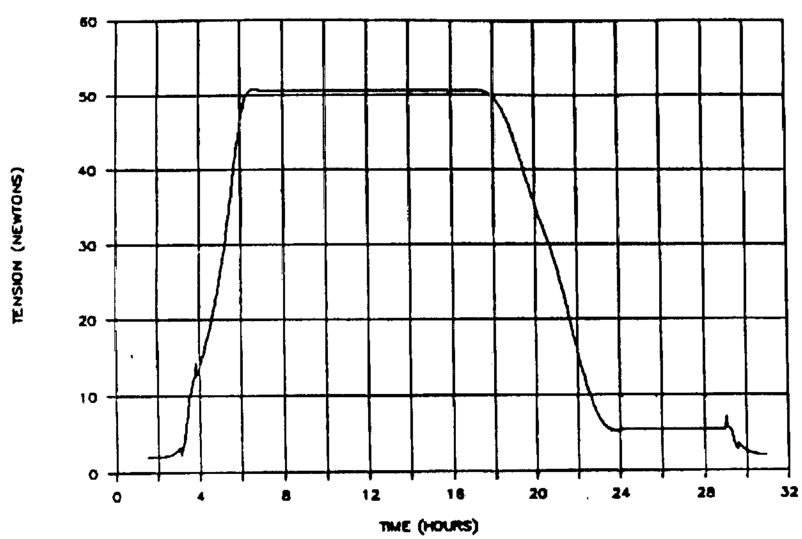
Nominal Mission Profile



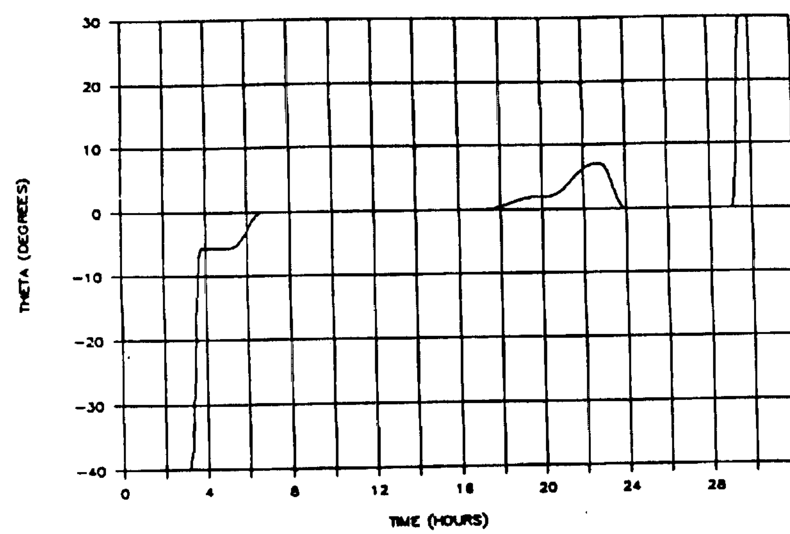
ℓ vs. Time for Baseline Reference Flight Profile



$\dot{\ell}/\ell$ vs Time for Baseline Reference Flight Profile



Tension (f) vs. Time for Baseline Reference Flight Profile



Seed Function for θ vs. Time for Baseline Reference Flight Profile

Orbital Dynamics

Soft Stop & Resume Profiles

The Deployer Control System Includes Soft Stop & Resume Profiles.

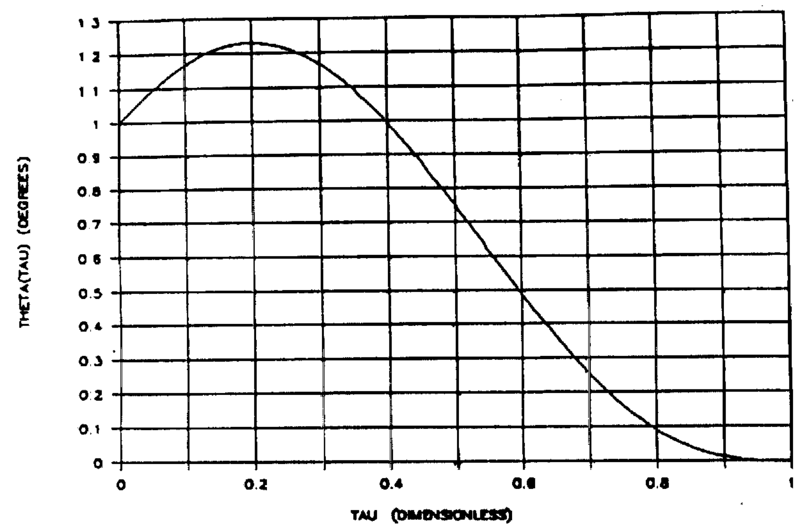
These Functions Allow the System to be Stopped During Deployment or Retrieval in a Controlled Manner without Sacrificing System Stability.

To Achieve this, there Needs to be Continuity in Commanded Length and Length Rate. The In-Plane Angle (q), and its Time Derivatives Must be Zero at the End of the Maneuver, and Must Match the Prevailing Values at the Beginning of the Manuever. From these Conditions Functions Were Developed to Schedule q Over the Manuever Segment.

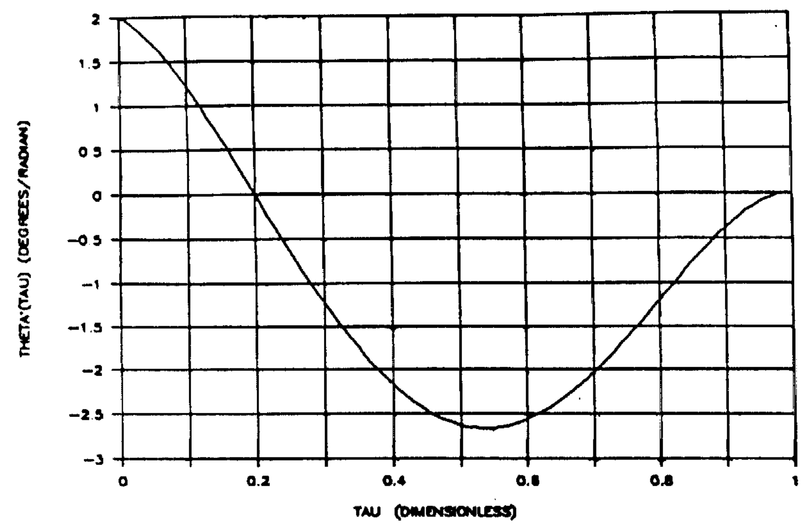
The Synthesized Alternative q Shape Functions are Shown on the Following Page.

Orbital Dynamics

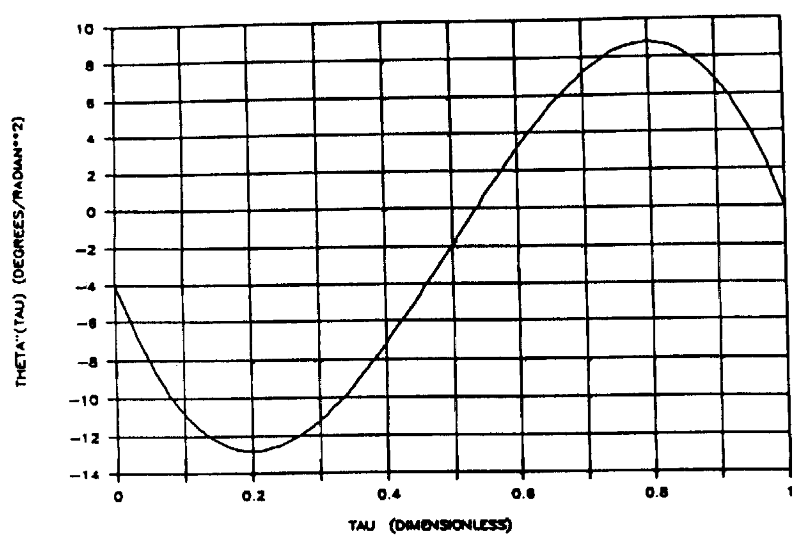
Soft Stop & Resume Shape Functions



Shape Function One for Soft Stop



Shape Function Two for Soft Stop



Shape Function Three for Soft Stop

Orbital Dynamics

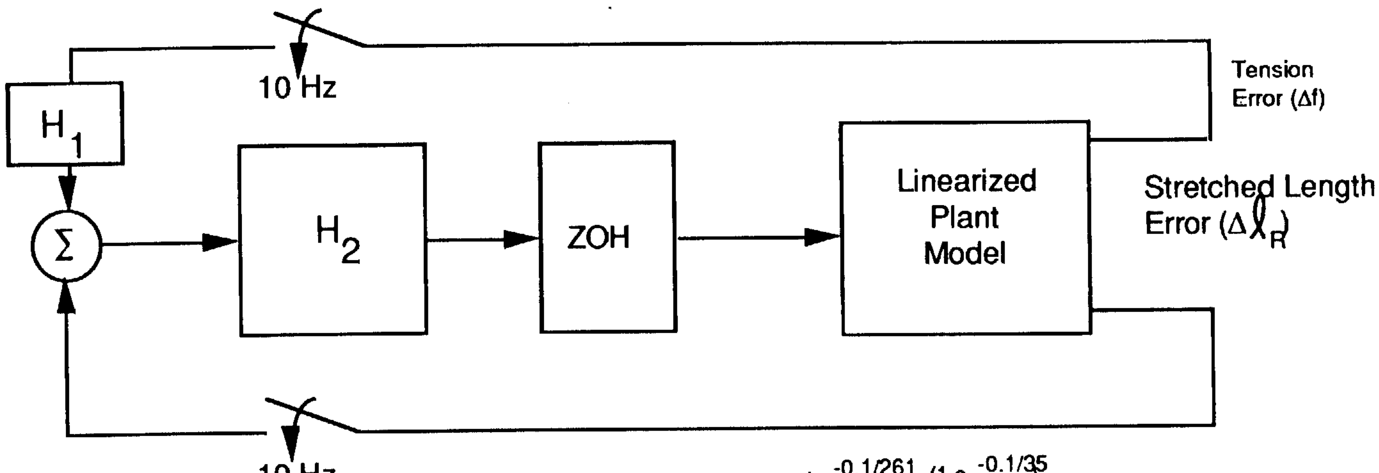
Deployer Control System Models

The Deployer Control System has 2 Modes, the Basic Mode Which Utilizes Tension Feedback, and the Proportional Mode Which Tracks the Tether Length as a Function of Time.

The Models for These Modes are Shown.

Orbital Dynamics

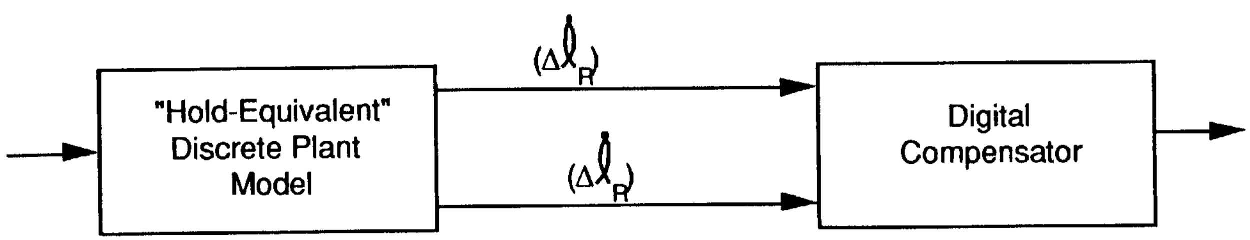
Linearized Control System Models



$$H_1 = \frac{-31.3}{0.157}\,\frac{1-e^{-0.1/700}}{z-e^{-0.1/700}}$$

$$H_2 = -0.157\,\frac{1-e^{-0.1/261}}{z-e^{-0.1/261}}\,\frac{(1-e^{-0.1/35})}{(z-e^{-0.1/35})}$$

Basic Mode Model



Digital Compensator Model:

$$\frac{0.00497508312542\,(1-R)}{(z-0.990049833749)}(\Delta\ell_R) + 40\,R\,(\Delta\ell_R) + 5.0\,R\,(\Delta\ell_R)$$

Proportional Mode Model

Orbital Dynamics

Comparison of Basic and Proportional Modes

General Characteristics of the 2 Deployer Control Law Modes are Given in the Table for Comparison.

The Basic Mode Will not Be Used for TSS-1.

Orbital Dynamics

Comparison of Basic and Proportional Modes

	Basic	Proportional
General Characteristics	Tether Length Varies Due to Tension Feedback Even When Constant Length is Commanded; Sensitive to Friction Effects; Tension Error from Predicted Value (Due to Sensor Error, Friction, Modelling Simplifications, Orbit Error, etc.) Causes Length Error.	Tether Length Insensitive to Tension, Tension Errors, and Friction; Accurate Tracking of Commanded Length; No Yo-Yo Damping of Libration
Damps Libration	Some (Limited by Friction)	No
Accurate Length, Length-Rate Control	No. Due to Lower Length-Loop Gain and Tension Feedback Component	Yes. Due to High Length Loop Gain, Lack of Tension Feedback Component
Usable Throughout Mission	Not Recommended	Yes

Orbital Dynamics

Controller Gain & Phase Margins from Random-Parameter Analysis

Gain and Phase Margins are Given for the Control Systems are Listed Below.

Orbital Dynamics

Controller Gain & Phase Margins from Random-Parameter Analysis

Conditions	Loop	Where Loop was Opened	Gain Margin (dB)	Phase Margin (deg)
On Station at 20 km	Basic	Tensiometer	3.29	18.43
		Motor	2.80	14.64
		Encoder	4.20	41.76
Full Mission, Original Gains	Proportional, Libration Not Modelled	Motor	4.43	45.77
Where Rate < 0.1 m/s, Lowered Rate-Mode Gains	Proportional, Libration Not Modelled	Motor	10.95	38.55

Orbital Dynamics

Operational Hazards Study Plan (OHSP)

Severed Tether Studies
- Severed Tether is an Improbable Occurance
- Critical Distances are a Function of Tension (Separation Distance)

Reaction Control System Studies
- Boom Loads are Not a Problem
- Orbiter Controllability is Not a Problem
- Satellite Retention in the Docking Ring is Not a Problem

Boom/Satellite Collision Studies
- Satellite Can Collide with the Boom at a Relative Velocity of 0.3 m/s with No Boom Damage

Boom Ejection Studies
- The Relative Trajectories Allow Sufficient Time for an Orbiter Avoidance Maneuver

Satellite Boom Ejection Studies
- No Orbiter Collision Problems

Sensor Error Study
- Reasonable Sensor Errors are Not a Problem for the Deployer Control System in General

Satellite Run-away & Overtorque Studies
- Can be Caused by Hardware or Software Failure
- Potentially the Most Hazardous Failure
- The Hazard is Controlled by Overspeed Protection
- No Boom Loads Problem

Orbital Dynamics

Operational Hazards Study Plan (OHSP)

OHSP Number	Title/Subject	Status
TSS-OHSP-04	Severed Tether Studies	Closed
TSS-OHSP-05/10	Reaction Control System Studies	Closed
TSS-OHSP-07/08	Boom/Satellite Collision Studies	Closed
TSS-OHSP-06	Boom Ejection Studies	Closed
TSS-OHSP-12	Satellite Boom Ejection Studies	Closed
TSS-OHSP-11	Sensor Error Study	Closed
TSS-OHSP-07/08	Satellite Run-away & Overtorque Studies	Closed

Orbital Dynamics

String Dynamics/Satellite Attitude Dynamics

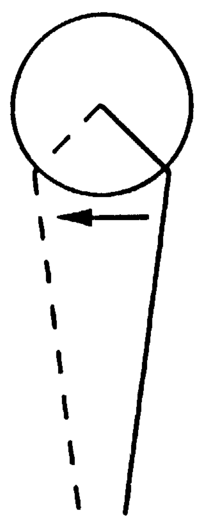
Satellite Attitude (Pendulous) Dynamics and String (Skiprope) Dynamics are Defined in the Figures Below.

Skiprope Can be Excited by Current Flowing in the Tether. Recent Studies Indicate that ≈ 35 m of In-Plane and 10 m of Out-of-Plane Mid-Node Tether Amplitude can be Excited.

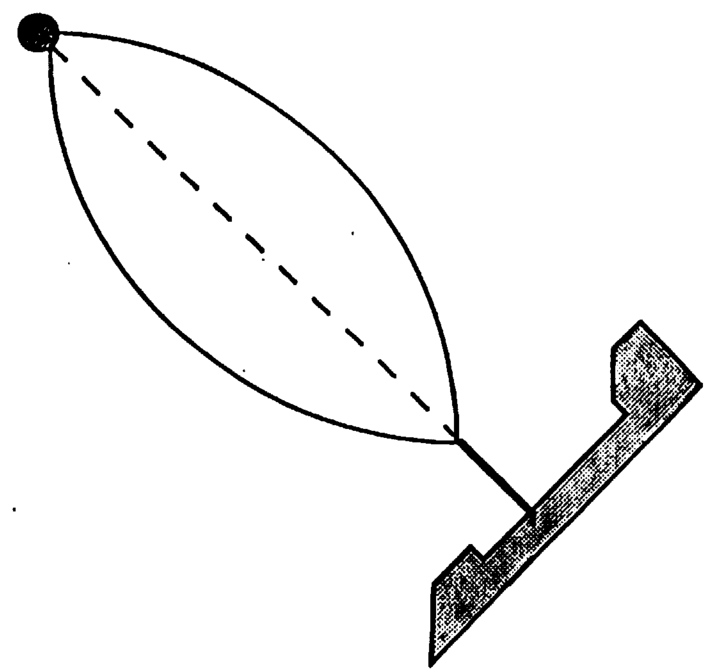
The On-Station 1 Tether Skiprope Amplitude of 35 m by 10 m Results in an On-Station 2 Circular Mid-Node Amplitude of ≈ 30 m

Orbital Dynamics

String Dynamics/Satellite Attitude Dynamics



SATELLITE ATTITUDE (PENDULOUS) DYNAMICS



STRING (SKIPROPE) DYNAMICS

Orbital Dynamics

String Dynamics/Satellite Attitude Dynamics

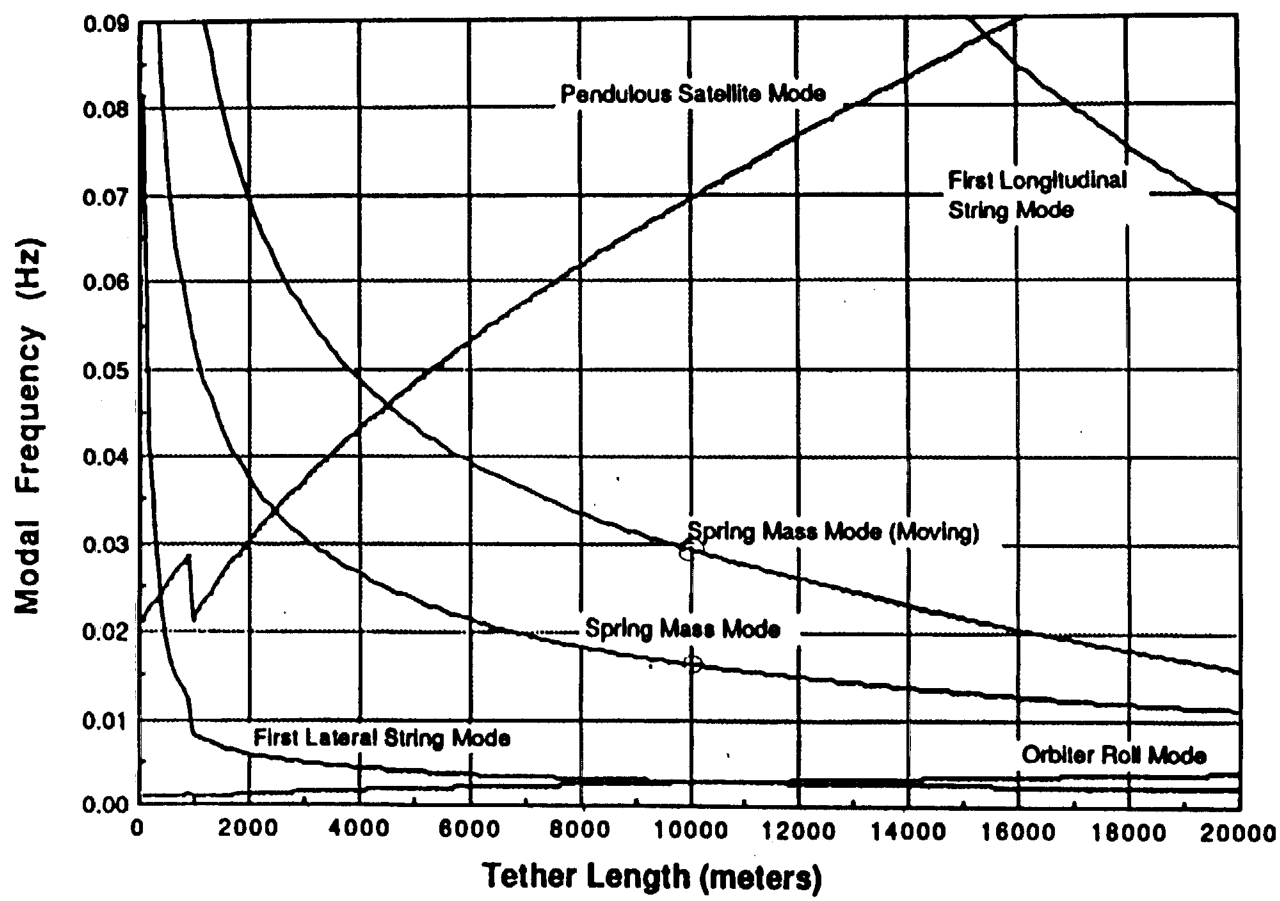
At Approximately 420 m the Satellite Pendulous and Skiprope Frequencies are the Same and Skiprope Energy is Transferred to the Satellite Attitude Motion. At this Range Approximately 6 Degrees of Satellite Pendulous Amplitude Results For Every meter of Skiprope Amplitude at 2.4 km.

Skiprope Amplitude Reduction Occurs at $\approx$ 4.5 km Due to the Coupling Between the Satellite Pendulous Dynamics and the Tether Bobbing Dynamics.

A Plot of the Frequency Content of the Dynamics in Variable Geometry Tether System is Shown on the Following Page.

Orbital Dynamics

Frequency Content of Tether Dynamics



Orbital Dynamics

Skiprope Studies

An Orbiter Yawing Maneuver with the Orbiter in a Local Vertical Orientation has Been Identified as an Effective Means of Skiprope Damping. A 35 m Mid-Node Amplitude can be Reduced to 3 m in About 10 minutes. Accurate Observation of Skiprope is Required for this Maneuver. The Maneuver is Shown Below.

Changing the Tension in the Tether at the Right Phase of Satellite Pendulous Motion Can Effect the Amplitude of Satellie Pendulous Motion. This Change in Tension can be Generated by Satellite Thruster or Orbiter PRCS $\pm$ Z Firings.

Reducing the Retrieval Rate Can Reduce an On-Station Amplitude 35 m by 10 m at 20 km to 10 m at 2.4 km.

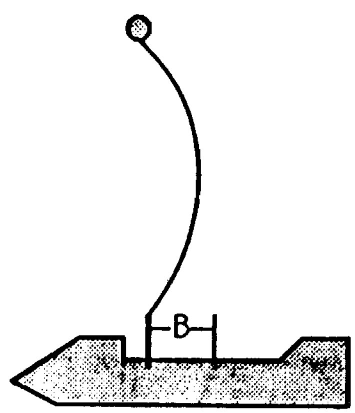
Reduction in Current Levels Yields Approximately Proportional Reduction in String Motion Amplitude.

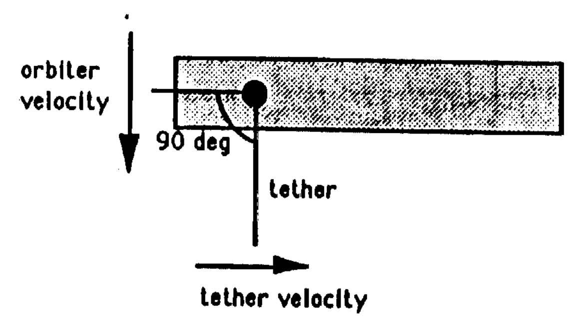
Current Flow in the Tether Can be Phased with the Skiprope Motion in a Manner Which Reduces Skiprope Dynamics.

A Skiprope Damping Mechanism has Been Designed to Eliminate Residual Skiprope Motion During the Final Phase of Retrieval Before Satellite Docking.

Orbital Dynamics

Orbiter Yaw Maneuver





Orbital Dynamics

Skiprope Observer

Some Methods for Reducing the Skiprope Amplitude Require that the Skiprope Motion be Observed so that Proper Phasing of Maneuvers and Current Flow Can be Achieved.

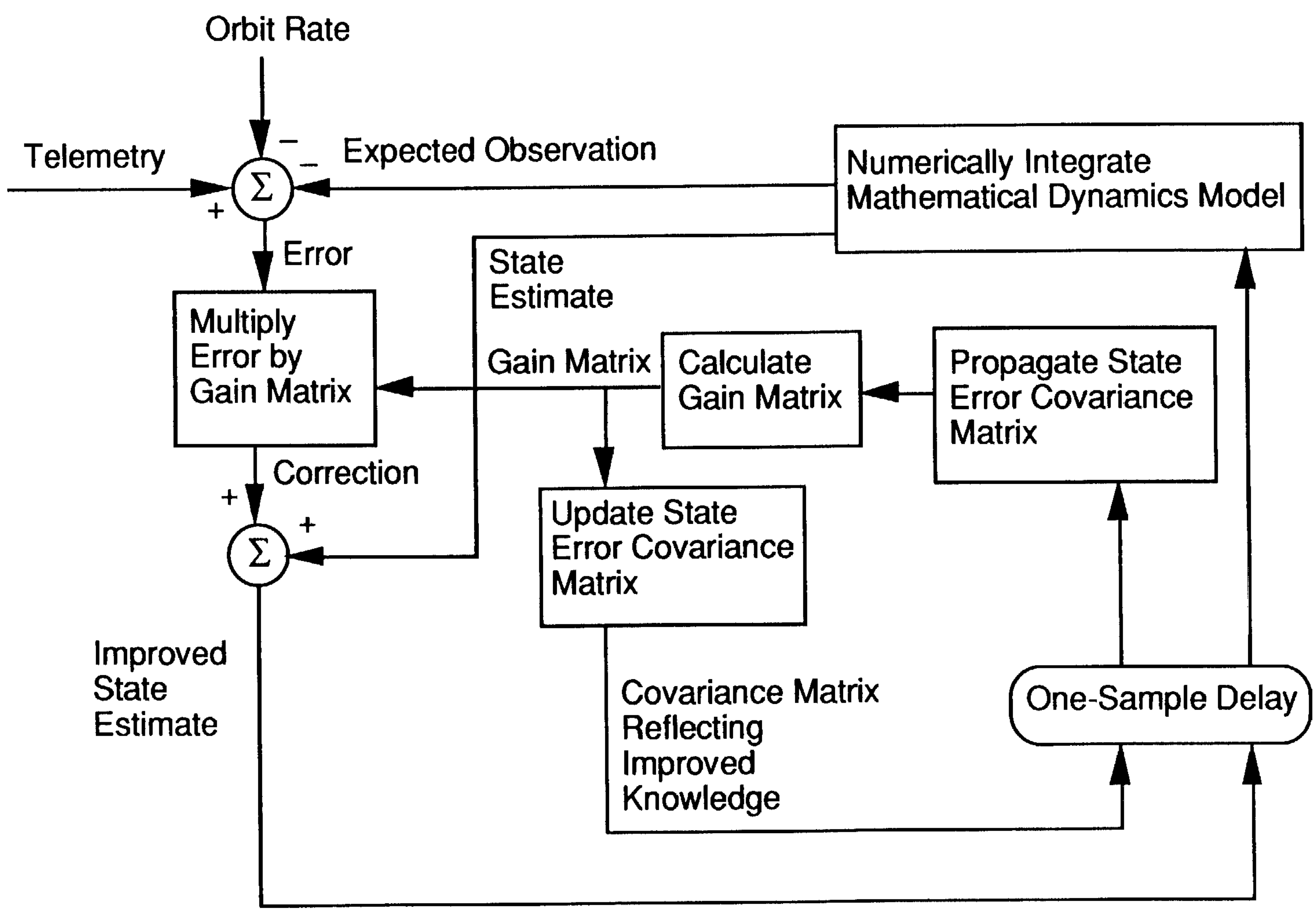
The Program Known as the Time Domain State Estimator or "Filter" is an Iterative Variant of Leastsquares Curve Fitting. As the Filter Operates, a Mathematical Model in the Filter Continually Predicts What the Next Telemetry Reading Will Be. The Filter Multiplies the Error in Each Prediction by a "Gain Matrix" and Feeds the Result Back to Adjust the Model's "State Variables," Gradually Forcing the Model to Match the Physical System.

In Addition the Filter Estimates the Accuracy of Its Predctions and of the Measurements Through Knowledge of the Sensor's Noise Characteristics. This Allows the Filter to Adjust the Gain Matrix so that Measurements from Accurate and Inaccurate Sensors Can Be Combined with Appropriate Emphasis. The Filter Also Automatically Accounts for the Fact that the Model Will Drift from Reality During Telemetry Dropouts. As a Result, It Emphasizes Measurements Received Immediately after a Dropout More than It Would if There were No Dropout.

Because the Filter's Output Does not Depend on a Single Sensor, It Can Continue to operate, with Reduced Accuracy, When One or More of the Sensors Fail.

Orbital Dynamics

Skiprope Observer Block Diagram



5. Satellite

Satellite

Introduction

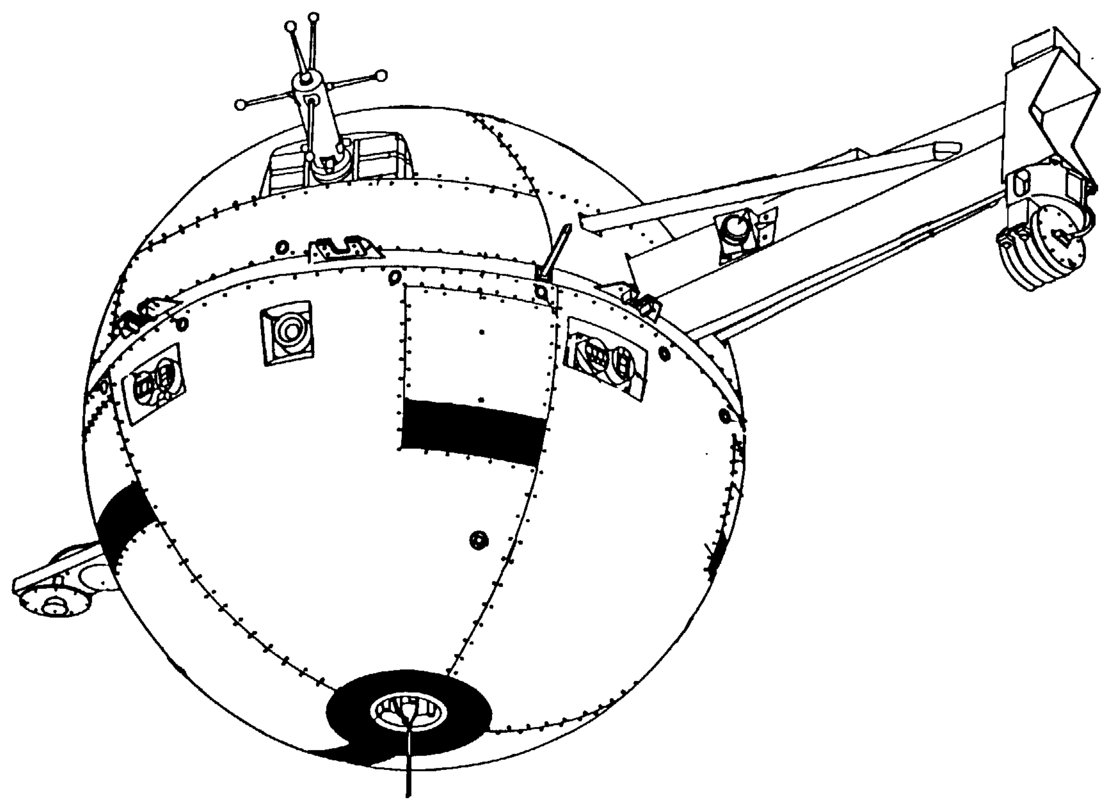
The TSS Satellite is a Spherical Multi-Mission Vehicle Designed to Carry Various Experiments with Different Mission Characteristics. the Satellite Has a Radius of 800 mm (31.5 in) and Weighs Approximately 513 Kg (1139 lbs).

The Satellite Consists of the Following Subsystems:

- Satellite Structure
- Electrical Power & Distribution System
- Onboard Data Handling System
- Telemetry Tracking & Control
- Auxiliary Propulsion System
- Attitude Measurement & Control System
- Thermal Control System
- Deployable/retrievable Booms
- Satellite Mounted Science

Satellite

Introduction



Satellite

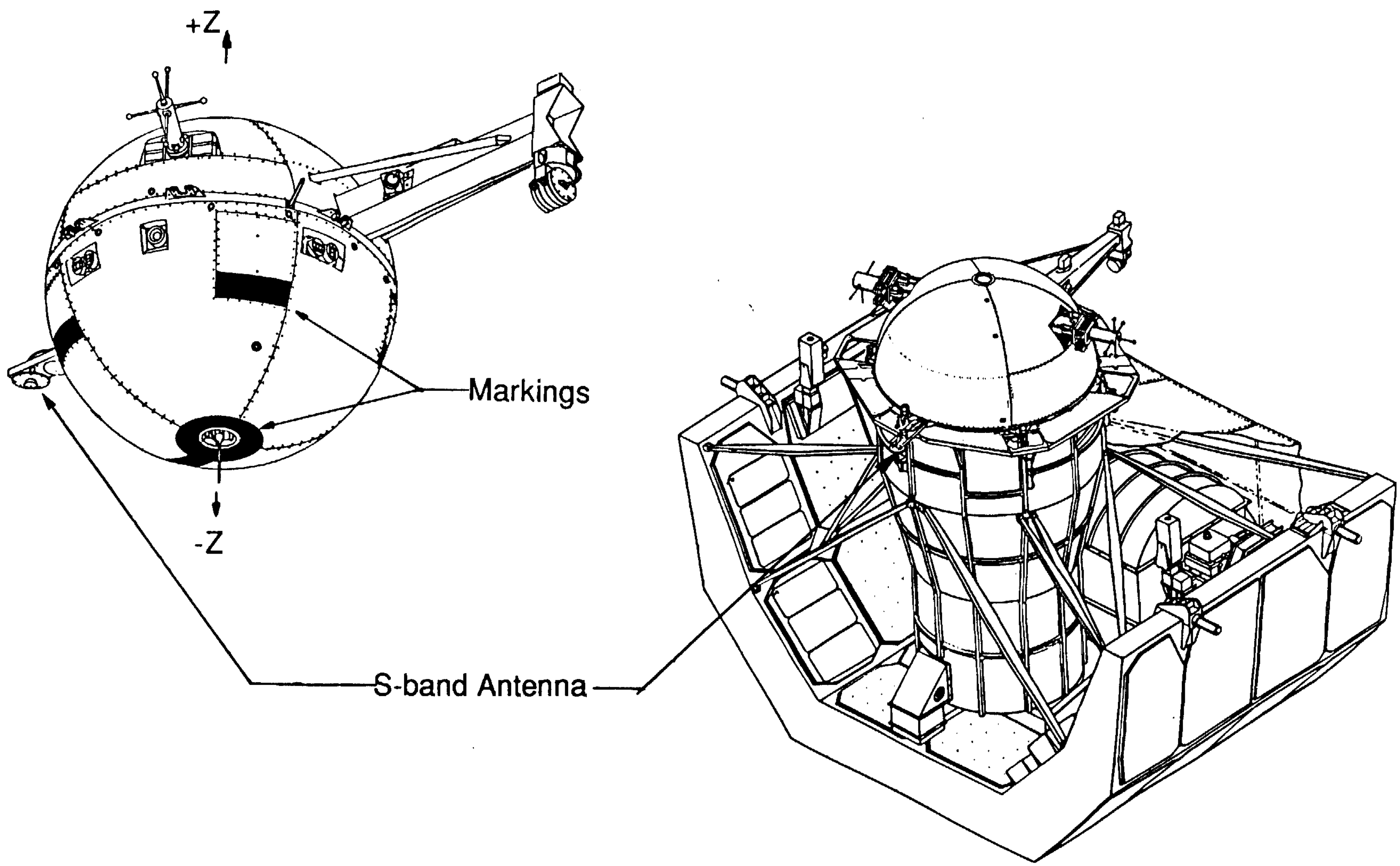
Orientation

When the Satellite is Latched to the Deployer 454 Ring, the S-band Antenna is 169° from Orbiter Flight Direction.

At Fly-Away and Prior to Satellite Spin Maneuvers, the Satellite S-band Antenna is Oriented in the Satellite Flight Direction.

Distinctive Black Markings Are Painted on the White Satellite Skin to Aid the Crew in Determining Satellite Attitude for Docking.

Satellite
Orientation



Satellite Structure

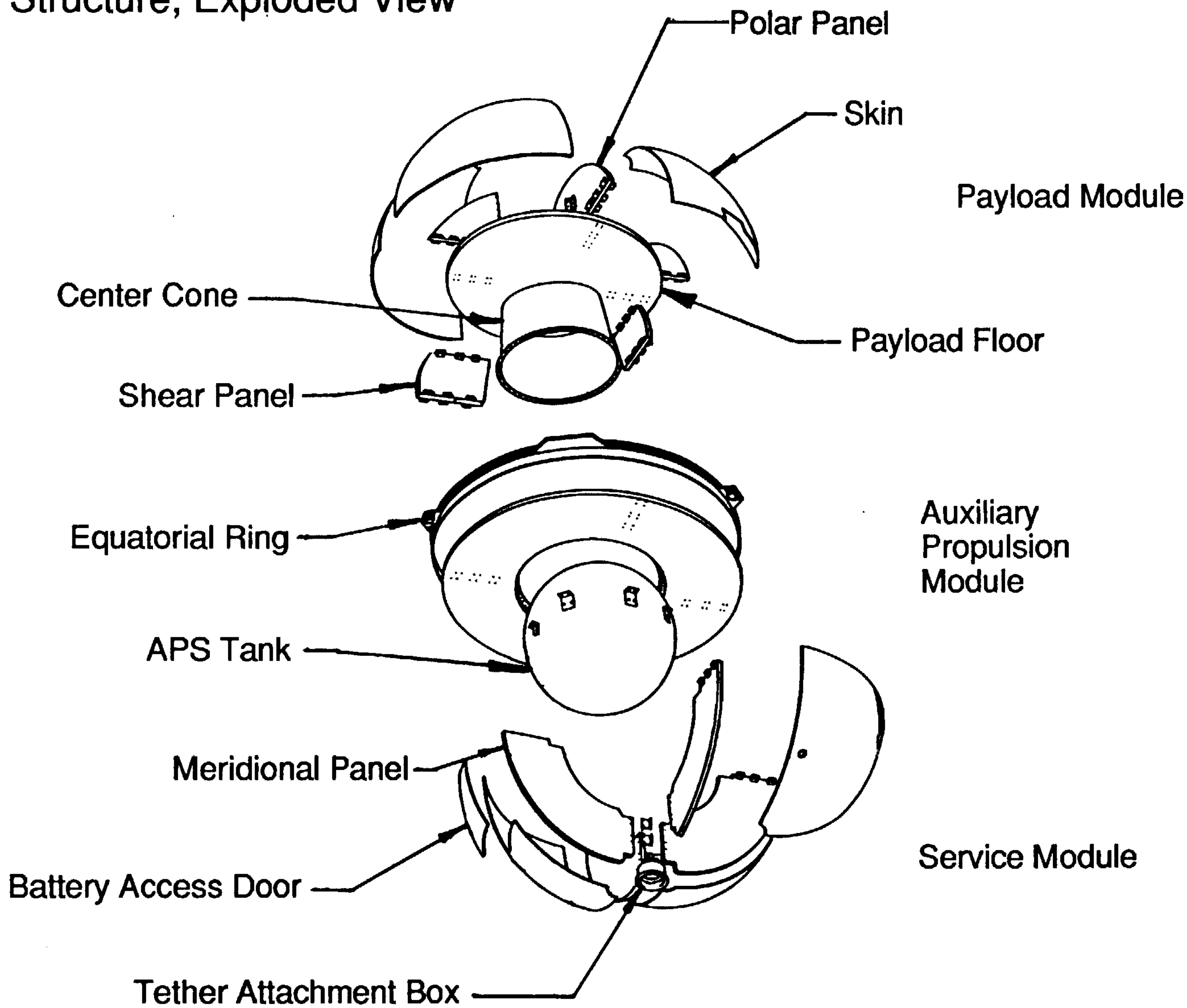
Overview

The Satellite Sphere is Structured into Three Modules:

- Service Module
- Auxiliary Propulsion Module
- Payload Module

Satellite Structure

Satellite Structure, Exploded View



Satellite Structure

Service Module

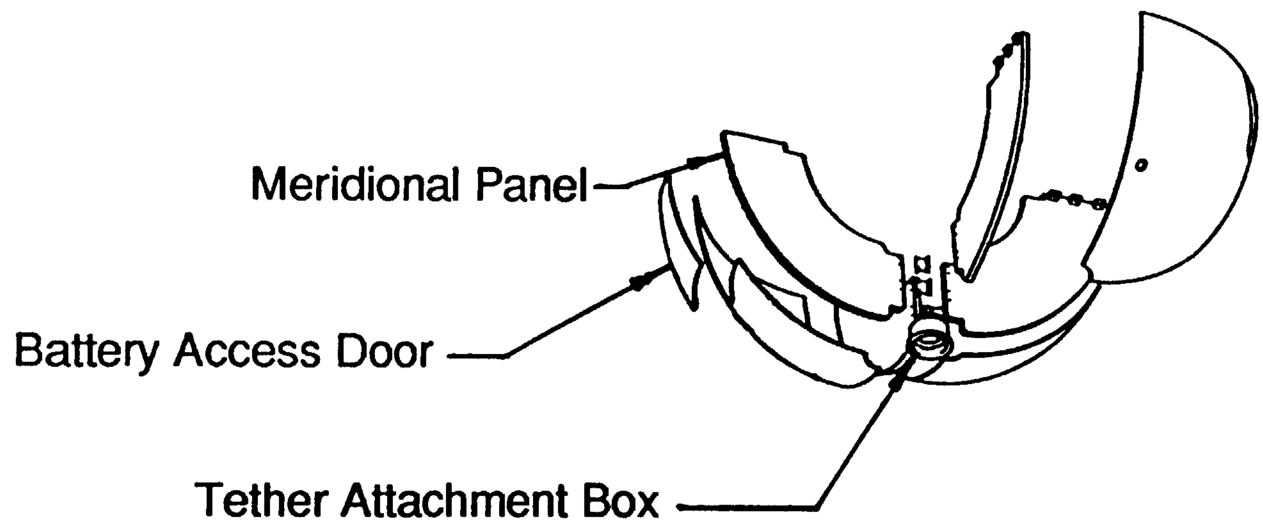
The Service Module is the Lower, or - Z, Half of the Satellite Sphere. It is Composed of 4 Meridional Panels and the Satellite Tether Attachment Box.

The Service Module Accommodates the Satellite Service Equipment Which is Mounted to the Meridional Panels. The Panels are a Honeycomb Structure Consisting of 2024-T81 Aluminum Skins and a 40 mm Thick Honeycomb.

The Satellite Skin is 1 mm Thick 2219-T62 Aluminum Which is 2 mm Thick at the Edges. The Skin is Attached to the Meridional Panels with NARMCO Cleats and is Thermally Decoupled from the Satellite Interior.

Satellite Structure

Service Module



Satellite Structure

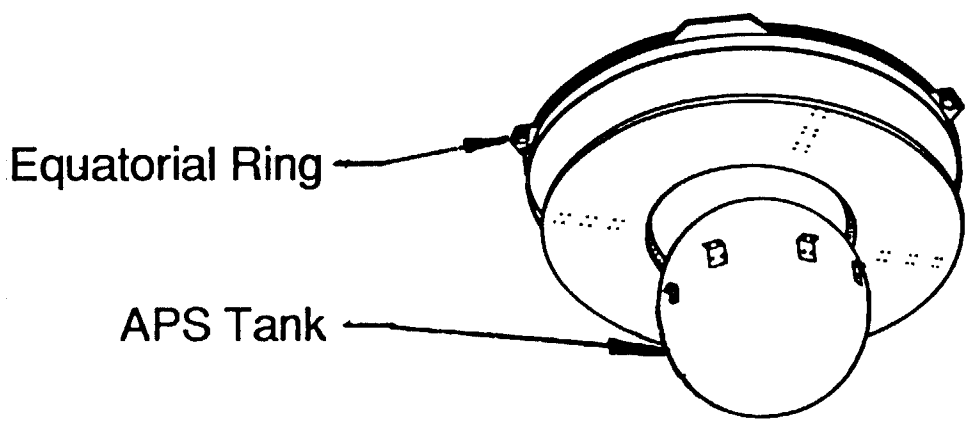
Auxiliary Propulsion Module

The Auxiliary Propulsion Module (APM) is Composed of the Equatorial Floor, the Equatorial Ring, the Auxiliary Propulsion System (APS), and the Earth Sensors (ES)

The Equatorial Floor and Ring Provide a Mounting Surface for the Satellite/Deployer Interface Hardware, the APS and the Earth Sensors.

Satellite Structure

Auxiliary Propulsion Module



Satellite Structure

Payload Module

The Payload Module (PM)is the Upper, or +Z, Half of the Satellite Sphere. It is Composed of 4 Honeycomb Shear Panels, 4 Honeycomb Polar Panels, a Honeycomb Center Core, a Honeycomb Payload Floor, and the Satellite Skin.

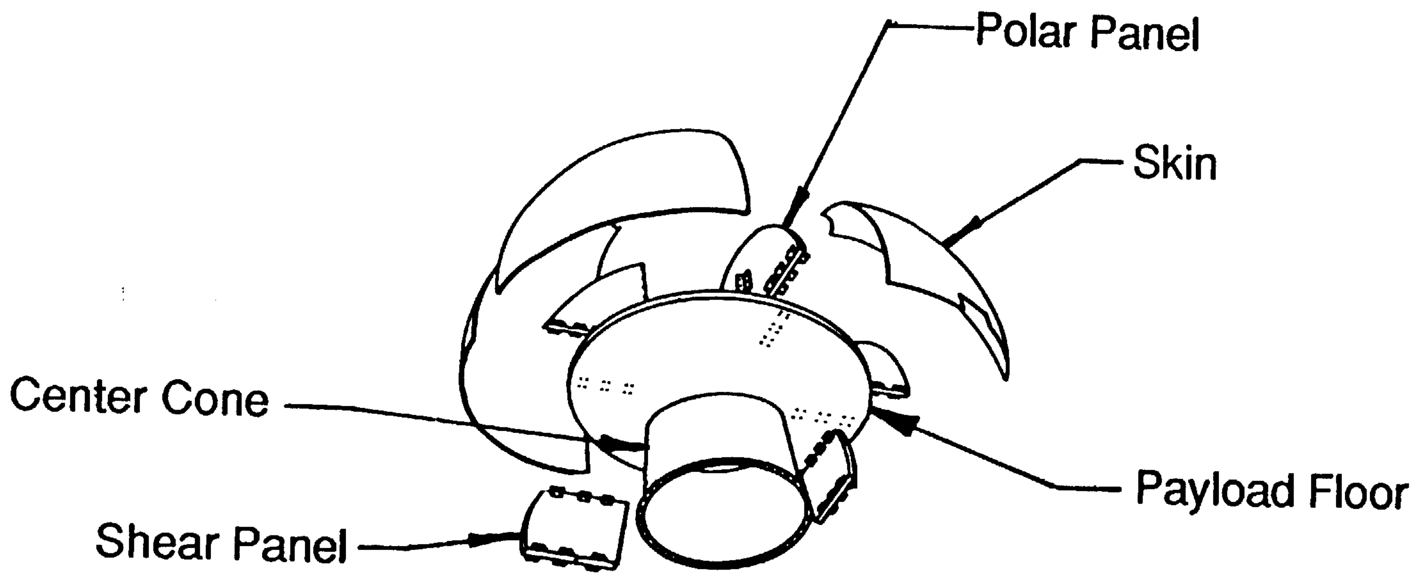
The TSS-1 Scientific Payload Equipment is Accommodated Within the PM.

Three Booms are Attached to the PM Structure, the Electrodynamic Mission (EDY) Boom, and 2 Deployable Retrievable Booms (DRB).

The Electrodynamic Mission Boom Consists of a Square Kevlar Tube and 2 Tubular Struts. This Boom Provides a Mounting Surface for Satellite Science Instruments.

Satellite Structure

Payload Module



Satellite Structure

Satellite/Deployer Interface

The Equatorial Ring is a 2219 Aluminum Ring with 6 Equatorial Ring Pads. It Provides the Structural Interface to Deployer.

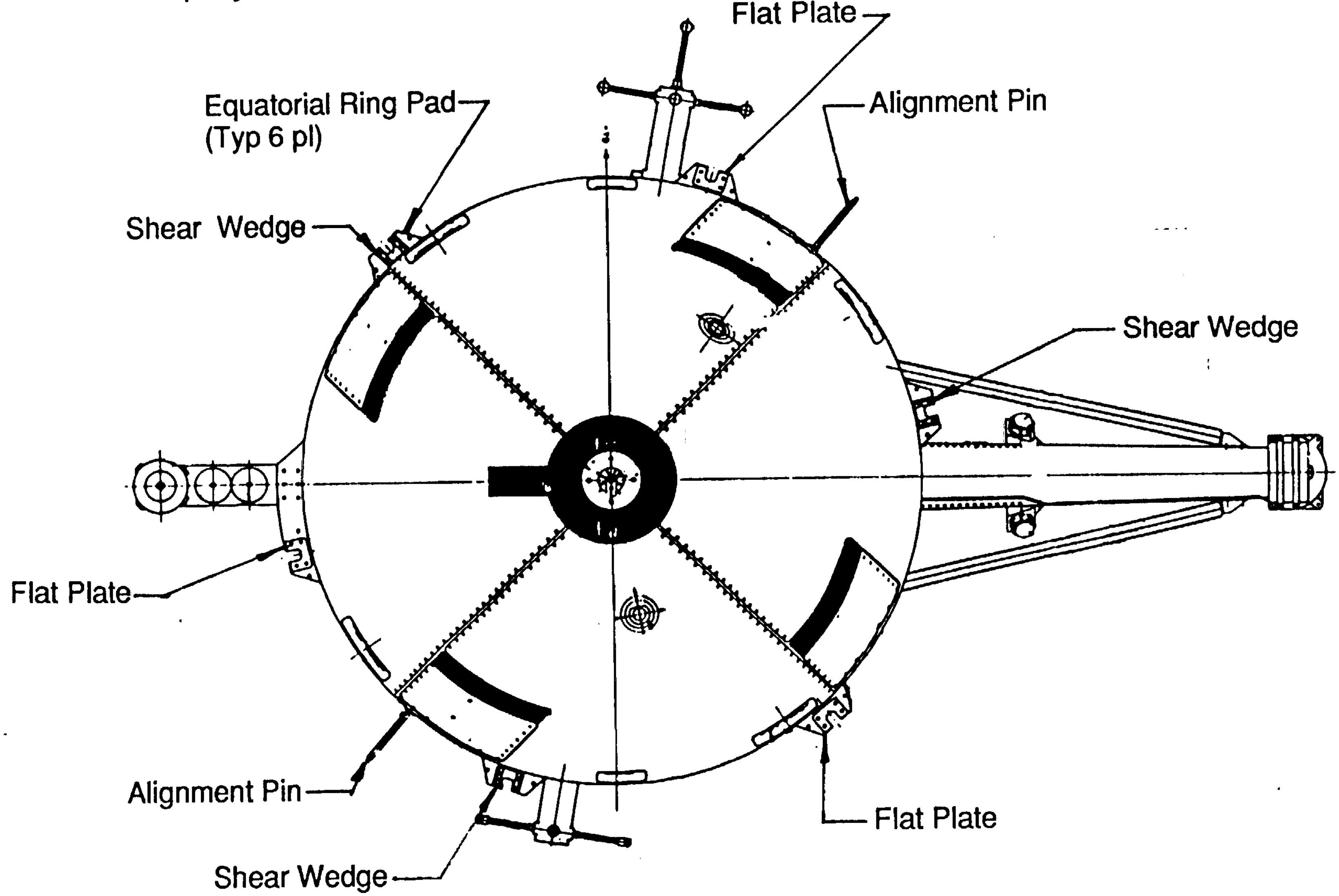
Mounted on -Z Side of Ring Pads are 3 Equally Spaced Shear Wedges and 3 Equally space Flat Plates. The Shear Wedges and Flat Plates are Made of Titanium for Thermal Isolation.

Mounted on the +Z Side of Ring Pads are 6 Striker Plates. The Deployer Satellite Restraint Latches Bear against These Plates

2 Alignment Pins are Mounted on the Equatorial Ring 55° from the Satellite X-Axis. The 2 Pins are 180° Opposed. They Consist of Aluminum Rods with Rollers on the Ends Which Level the Satellite during Fine Alignment.

Satellite Structure

Satellite/Deployer Interface



Satellite Structure

Satellite/Deployer Interface

The U1 Umbilical Connector is an ITT - Cannon Type Separable Connector. It is Demated by a Mechanism on the Deployer Prior to Deployment. The Connector is Mounted on a Bracket on a Meridional Panel in Service Module Below the Skin.

The U2 Umbilical Connector is Also an ITT - Cannon Type Separable Connector. It is Demated by a Deployer Mechanism. The Connector is Mounted in the Service Module Similar to the U1 Connector.

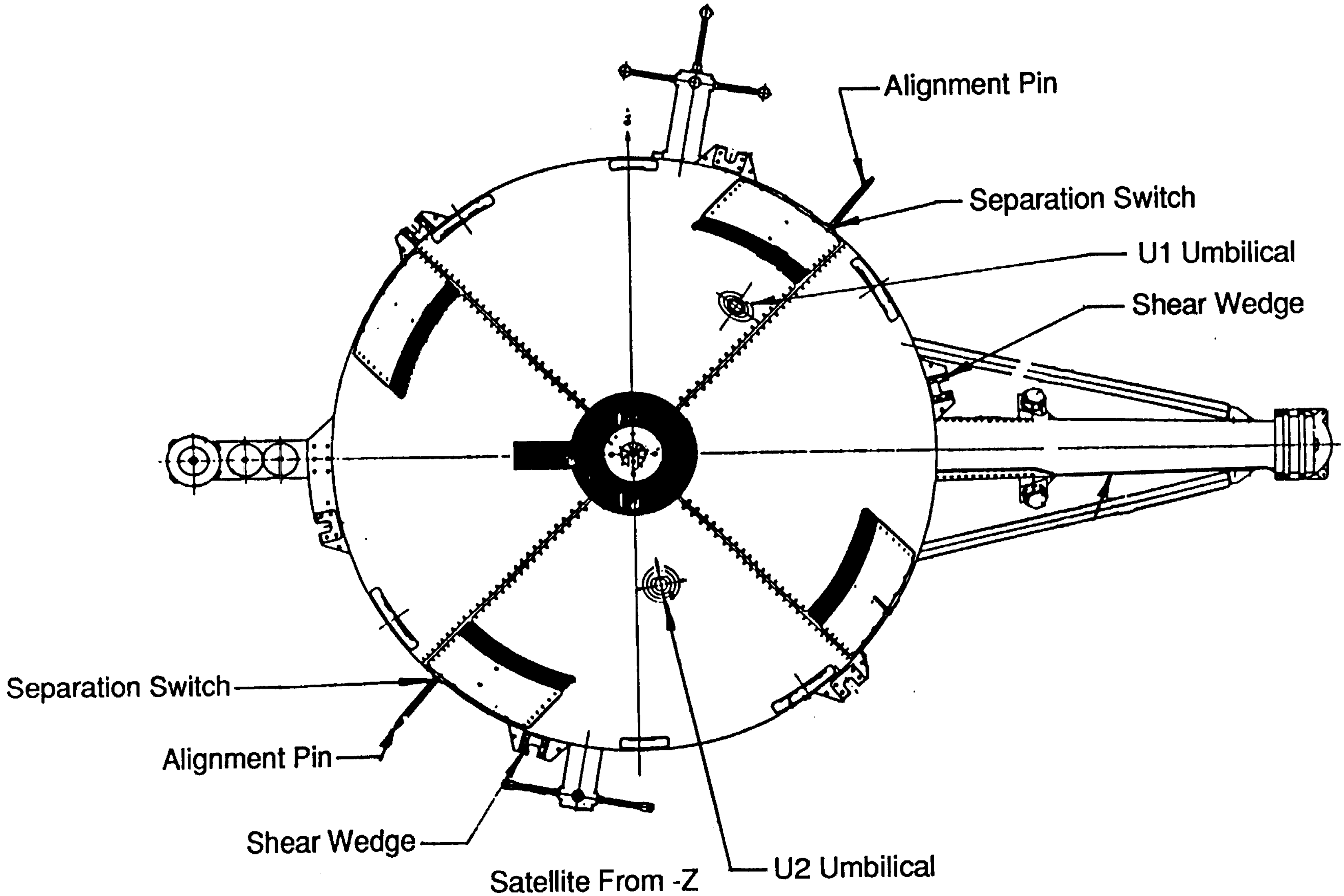
The Satellite Contains 4 Separation Switches which are Mounted in the SM, 90° Apart, 4° Below Equatorial Plane. One Pair is Mounted Below the Alignment Pins.

The Separation Switches Actuate Switches That Enable the Internal Satellite Batteries. The Switches are Wired with 2 in Parallel for Each of the 2 Battery Circuits.

The Separation Switches are Actuated by Unlatching the Deployer SRLs. The Satellite is Lifted 19.8 mm by the Deployer Plungers When the SRLs are Opened. The Separation Switch Actuator Ball Rides on a Ramp Cut into the SSA Structure. The Ball Extends (Closes) When the Satellite is Unlatched Retracts (Opens) When Latched.

Satellite Structure

Satellite/Deployer Interface



Satellite From -Z

Satellite Structure

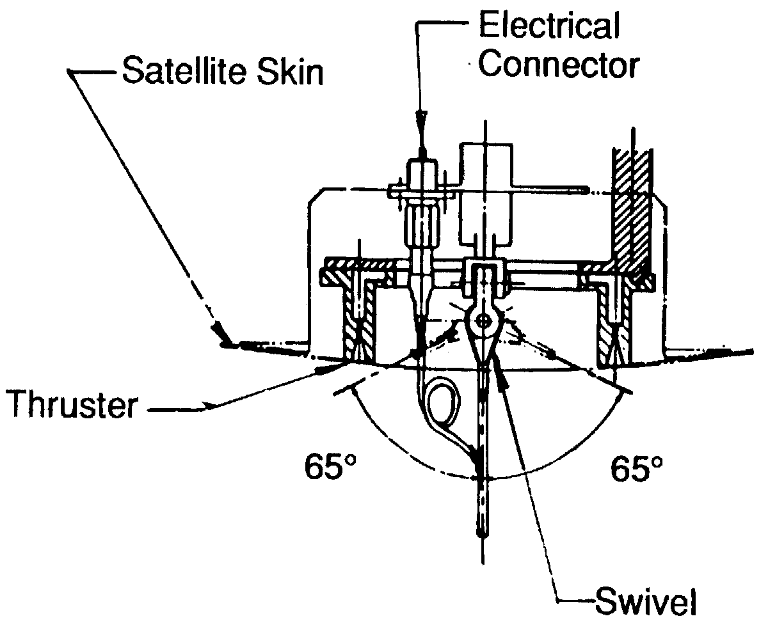
Satellite/Deployer Interface

The Tether Attachment is Mounted in a Pocket at the South Pole of the Satellite. It is Articulated for 2 Degrees of Freedom with a Universal Joint and Will Not Rotate Around the Z Axis.

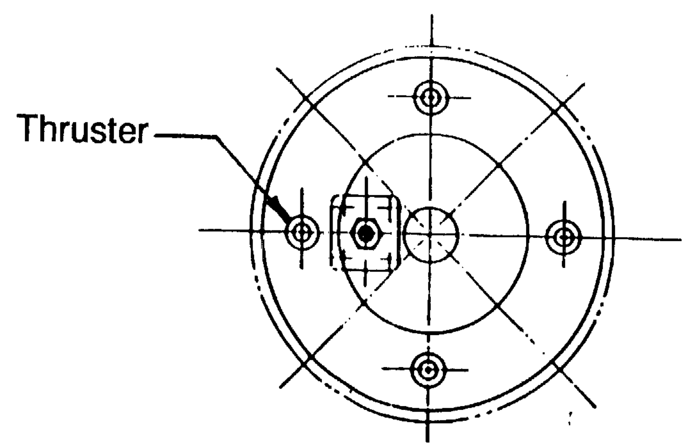
A High Voltage Electrical Connector Mounted Next to the Mechanical Attachment in the Bulkhead Fitting on the Satellite. The Connector is Lockwired to Satellite Structure During Assembly.

Satellite Structure

Satellite/Deployer Interface



Elevation



Plan
From -Z

Electrical Power & Distribution System

Overview

The Electrical Power and Distribution System (EPDS) Supplies Power to the Entire Complement of Satellite Components. Before U1 Umbilical Separation, Satellite Power is Provided by the Orbiter through the MPC and Satellite heater Power is Provided by the Orbiter through the MCA. After U1 Separation, All Satellite Power is Provided by Four Satellite Batteries.

The EPDS Consists of the Batteries, the Power Control and Distribution Assembly, and the Separation Switches and Power Verification Lamp Assemblies.

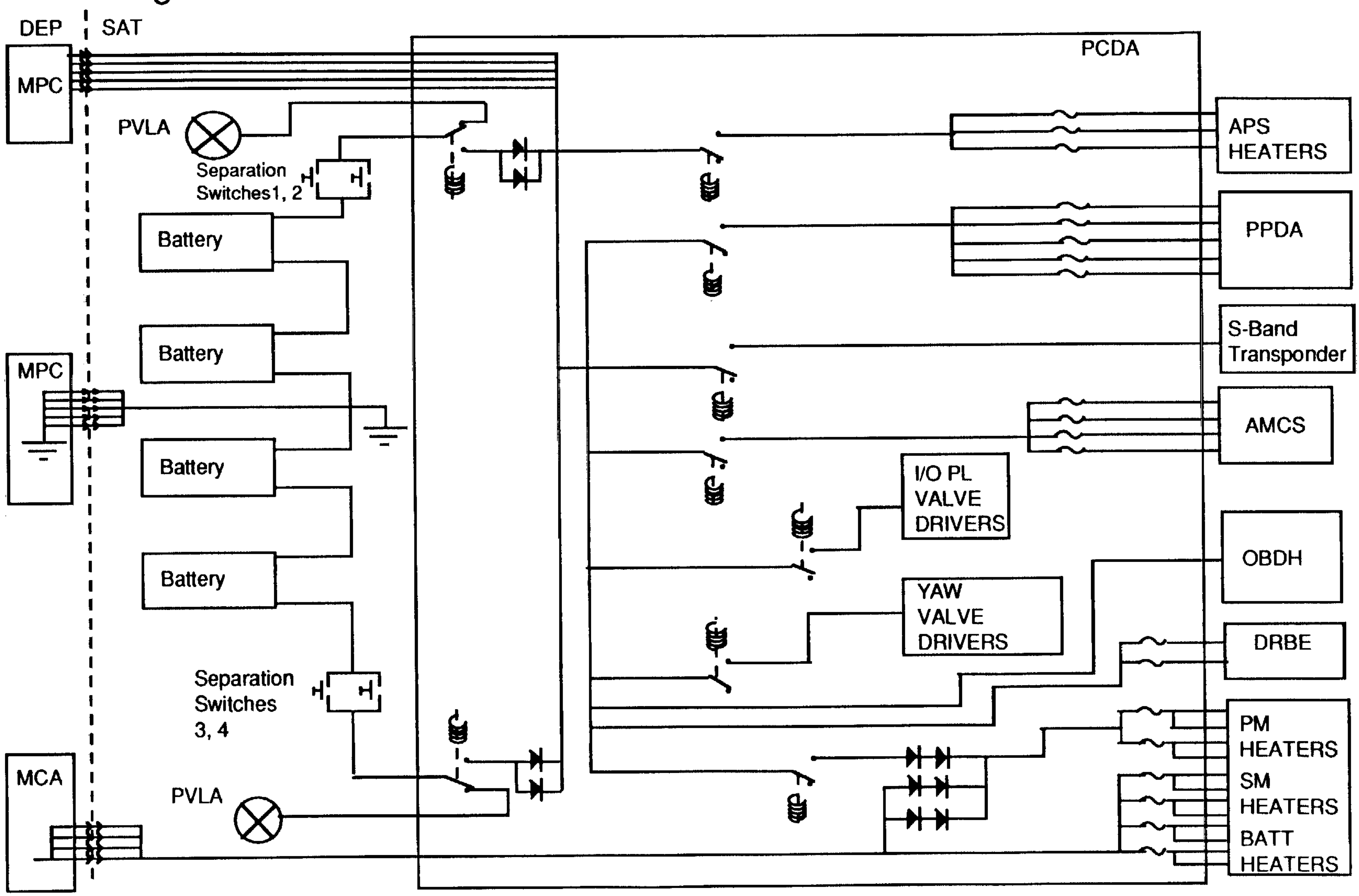
There are 4 Silver-Zinc Batteries in the Satellite. These Batteries are Mounted on the Meridional Panels in the Service Module. The Batteries are Connected as 2 Parallel Strings of 2 Batteries in Parallel. This Provides 2 Power Sources for the Satellite Main Bus.

The Batteries Can Deliver 7600 to 9200 Watt-Hours Each, with a Voltage Range of 14 to 19 Volts. The Batteries Have a Charged Life 35 Days. The Batteries Are Activated 4 Days Before Launch.

There are 2 Redundant Heaters on Each Battery Which are Enabled during Attached Phases. The Heaters are Powered through the U1 Umbilical.

Electrical Power & Distribution System

Block Diagram



Power Control & Distribution System

Power Control & Distribution Assembly

The Power Control and Distribution Assembly (PCDA) Provides Power Distribution to All Satellite System Components.

There is 1 Main Distribution Bus (Main Bus) Which Supplies Power to 10 System Distribution Busses. 8 of the Busses Are Fused. The On Board Data Handling (OBDH) and Telemetry Tracking and Command (TT&C) Busses Are Not Fused in PCDA

Power is Provided for Valve Drivers and Relay Drivers. Relays Enable the Battery String Connection and Disconnection. There are 2 Battery-String-on Drivers and 2 Battery-String-Off Drivers. A Capacitor Bank Ensures Turn-Off Power for the Batteries.

7 Distribution Busses Have Power Relays. The OBDH, DRB, and Relay/Valve Drive Have No Power Relays.

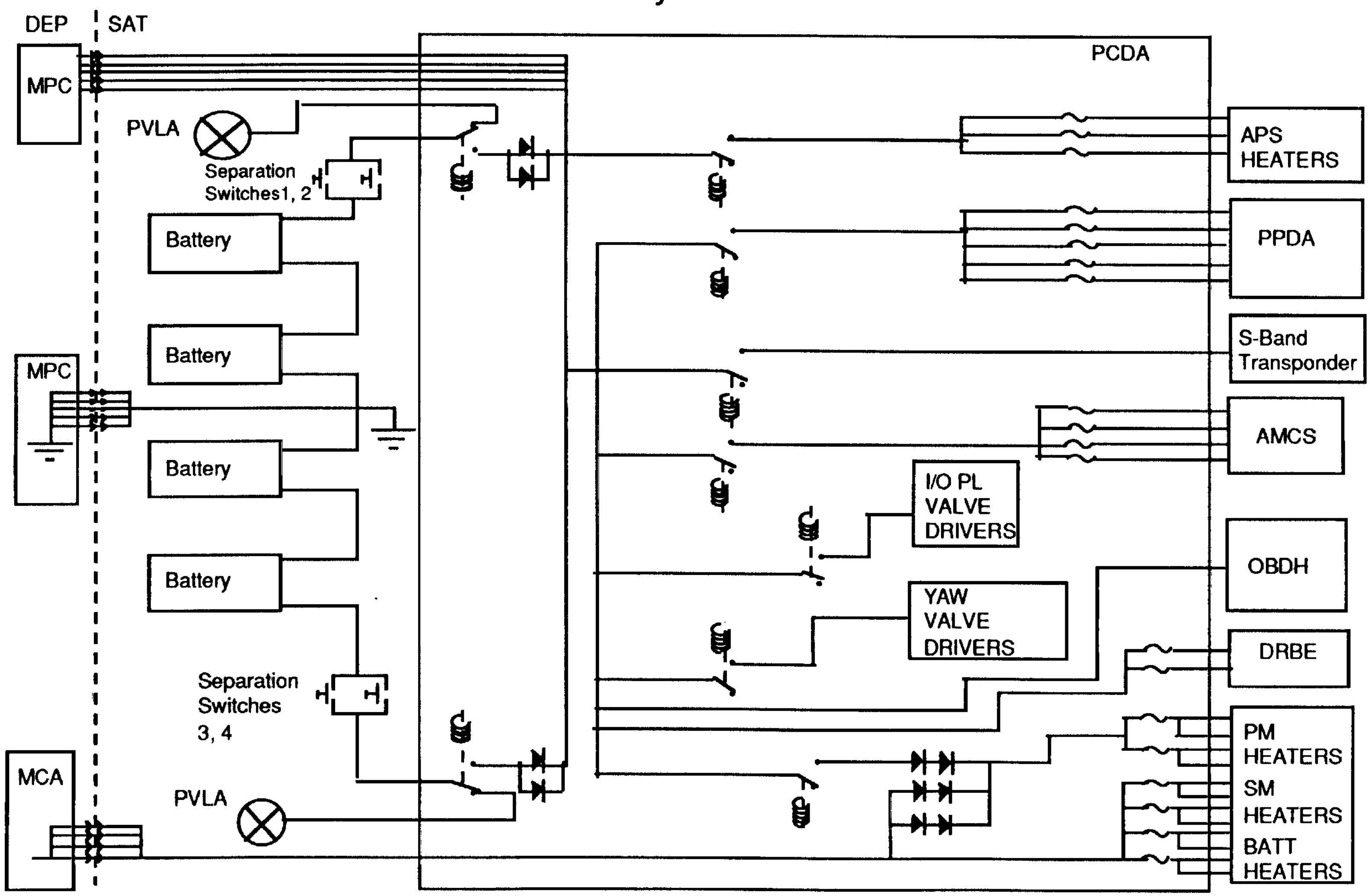
External Power is Provided through the U1 Umbilical. The Batteries Are Diode Protected.

The PCDA Interfaces with the OBDH Remote Terminal Unit Service (RTUS). Telemetry is Provided for Battery String Current, the 10 Distribution Bus Currents, and Power Control Relay Status.

The PCDA Contains Relay Drivers Which are Commanded by the OBDH RTUS and through the U1 and U2 Umbilicals. U1 Commands Include: Satellite Internal Power ON, Satellite Relay Reset, and Satellite Critical Function Relay Reset. U2 Commands Include Satellite Internal Power OFF.

Power Control & Distribution System

Power Control & Distribution Assembly



Power Control & Distribution System

Power Verification Light Assemblies

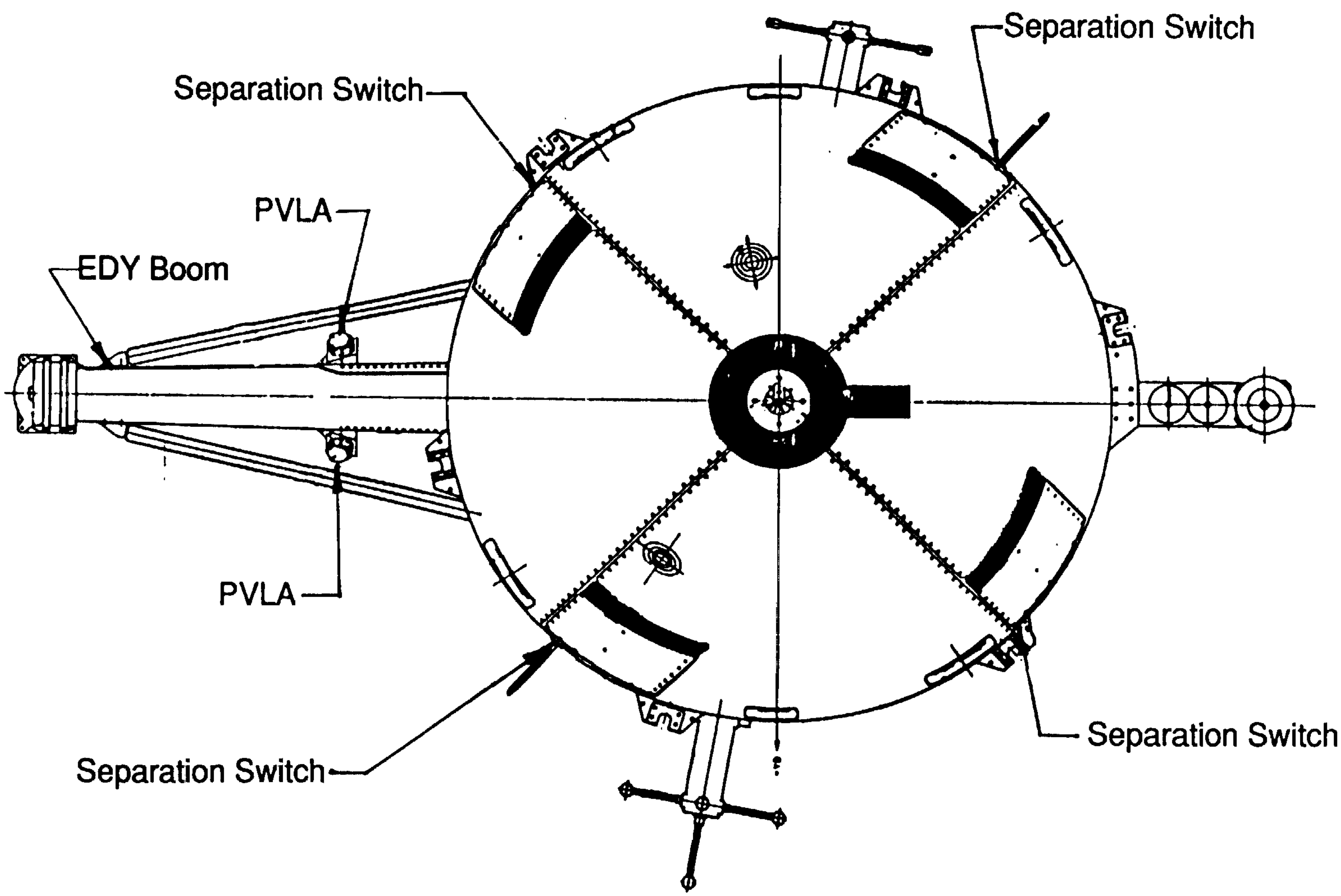
The Satellite Separation Switch System Includes 2 Power Verification Light Assemblies (PVLAs) to Provide Verification of Separation Switch Status and Main Bus Power Status. The PVLAs Are Mounted on the EDY Boom.

The PVLAs Emit 0.5 Candle of White Light Each and Consume 22 Watts Maximum Each. They are Treated with Magnesium Oxide to Minimize Reflected Sunlight

When the Battery Relay is Open (Internal Power OFF), the PVLA is Enabled. When the Separation Switch is Closed, the PVLAs Are Lit. When the Separation Switch is Open, the PVLAs Are Not Lit. The PVLAs Indicate That the Main Bus is Not Powered Via the Separation Switch and Via the Battery Relay.

Power Control & Distribution System

Power Verification Light Assemblies



Satellite From -Z

On Board Data Handling System

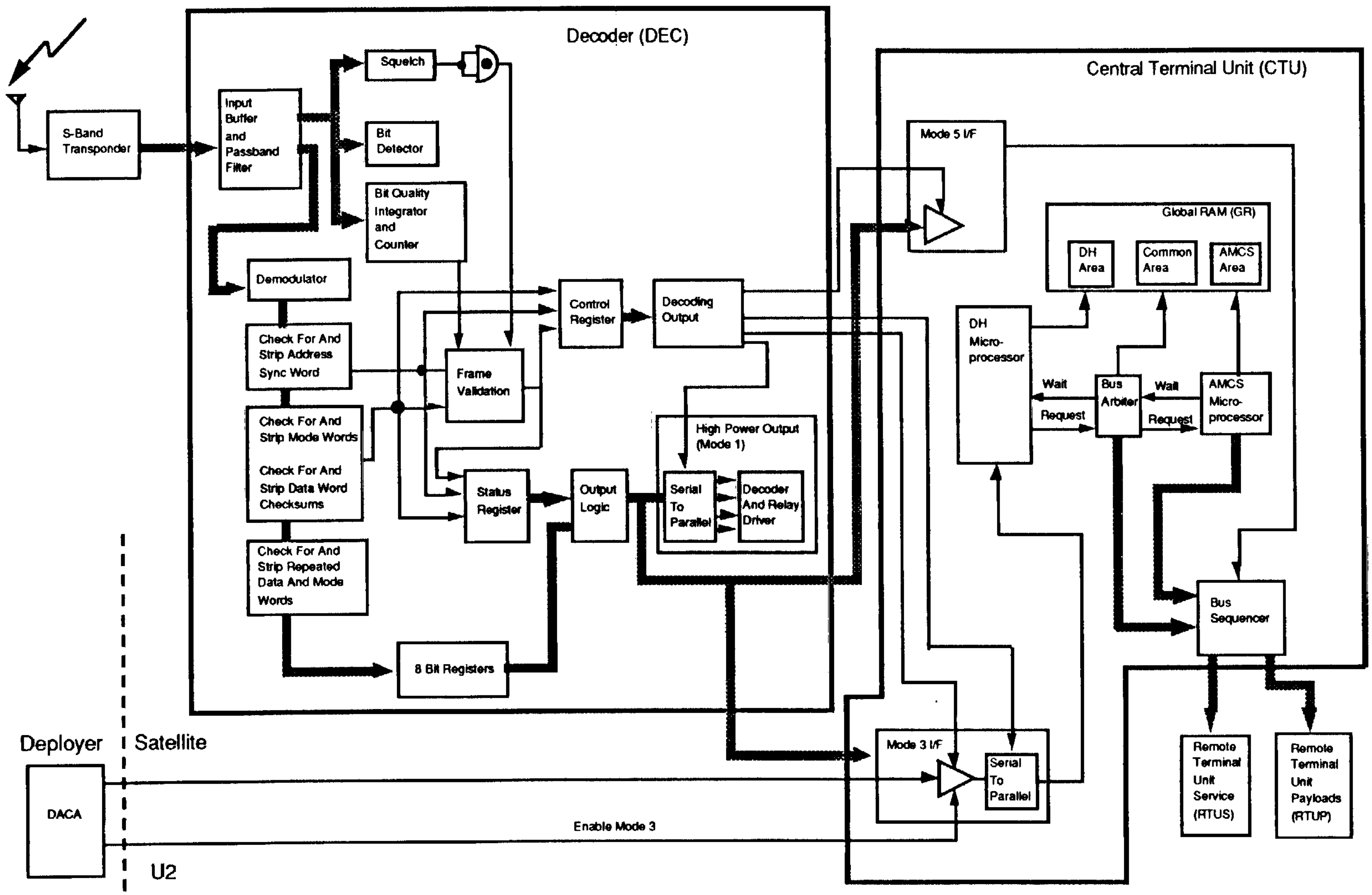
Introduction

The On Board Data Handling (OBDH) System Performs All the Data and Command Processing Tasks Within the Satellite. The OBDH is Modular So it Can Be Reconfigured to for Different Missions. Components Include:

- Central Terminal Unit (CTU)
- Memory Bank Unit (MBU)
- Decoder (DEC)
- Remote Terminal Unit (RTU)

On Board Data Handling System

Introduction



On Board Data Handling System

Central Terminal Unit

The Central terminal Unit (CTU) is the Central Controller of the OBDH. It Contains the Data Handling (DH) and AMCS Microprocessors. Each is a Hybrid Z-80 Microprocessor CPU with 2k bytes of RAM.

The DH Microprocessor Performs Command and On Board Data Management Tasks. The AMCS Microprocessor is Dedicated to AMCS Algorithms. Both Microprocessors Are Controlled by a 128 msec Duration Clock

The DH and AMCS Microprocessors Communicate through the Dual-Port Global RAM (GR)/ Bus Arbiter (BA). There are 3 Defined Memory Areas, the DH Area the Common Area, and the AMCS Area. Each Microprocessor Can Read from All 3 Areas, But Can Write to Only Its Area and the Common Area

The DH Microprocessor Receives All Commands and Writes the Data Required by the AMCS to the Common Area Where AMCS Microprocessor Can Read It. The AMCS Microprocessor Writes Data for Telemetry into the Common Area Which is Accessible in the Common Area by the Telemetry Format Generation (TFG).

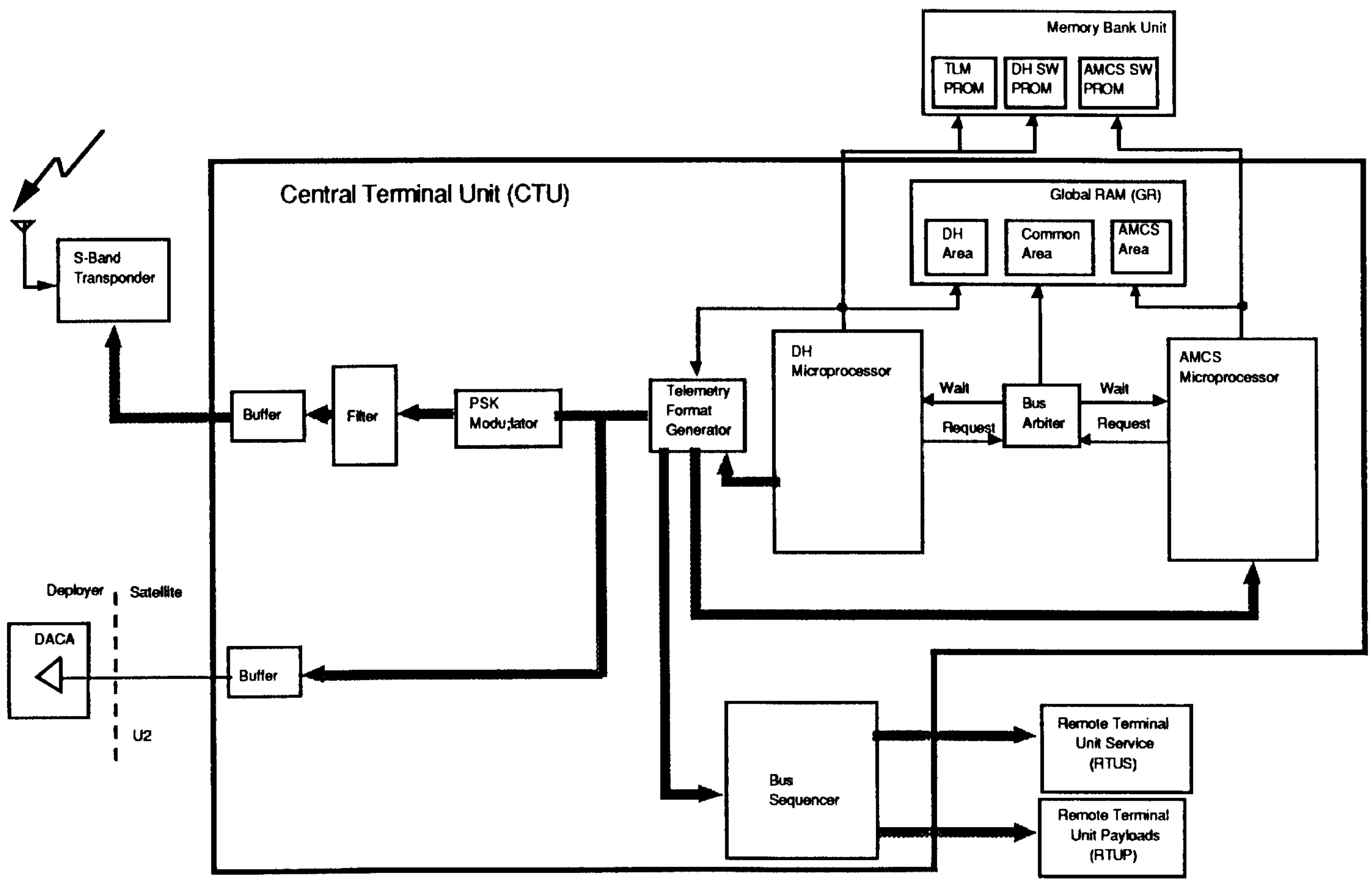
If a Power Drop-Out Occurs, the GA is Designed to Maintain Proper Voltages for >100 msec.

RAM Access Priority is Controlled by the BA.

The Memory Bank Unit Contains the Software Codes and Interrogation Instructions the CTU Uses When Building Telemetry. It Contains Up to 40 Kbytes of Program Memory.

On Board Data Handling System

Central Terminal Unit



On Board Data Handling System

Decoder & Remote Terminal Units

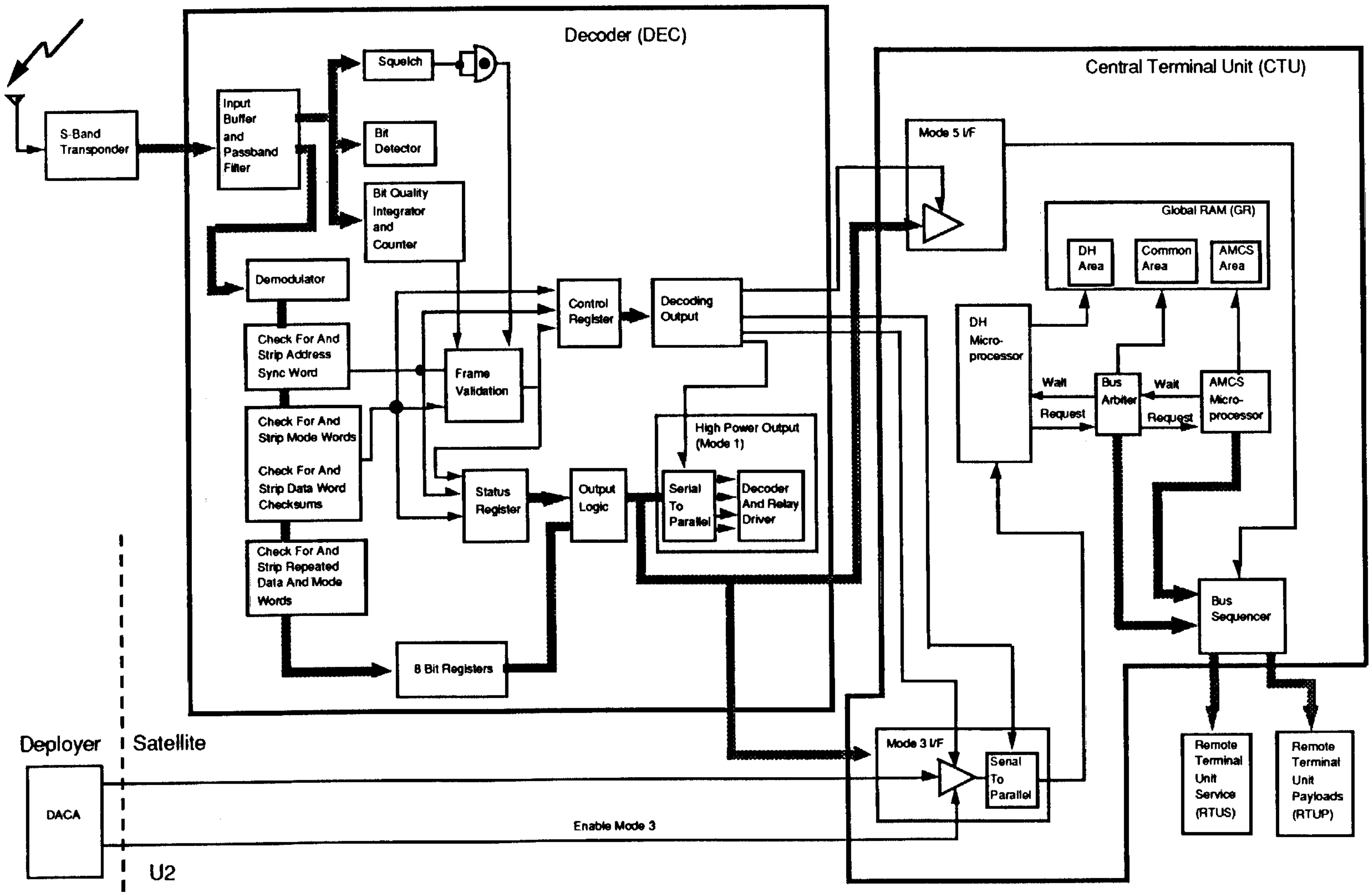
The Decoder Converts the PSK Modulated Input from the Receiver into a Non-Return-to-Zero Level (NRZ-L) Serial Command Signal.

There Are 2 Remote Terminal Units (RTUs). The Remote Terminal Unit Service (RTUS) is Dedicated to All Satellite Housekeeping Hardware Interfaces. The Remote Terminal Unit Payloads (RTUP) is Dedicated to all Experiment Hardware Interfaces.

The Hardware I/O Terminal of the OBDH Data Handling (DH) Task RTU Acquires Data from the Users and Distributes Commands and Special Signals. The DH Microprocessor Communicates with Either the RTUS or RTUP. The AMCS Microprocessor Can Communicate with Only the RTUS.

On Board Data Handling System

Decoder & Remote Terminal Units



On Board Data Handling System

OBDH Command/Telemetry Processing

There Are 3 Command Formats, Mode 1, Mode 3, and Mode 5.

Mode 1 Commands are High Priority Commands Routed Directly to the Appropriate Relay or Valve Driver after Processing by the Decoder. These Include Relay ON/OFF Commands or Valve OPEN/CLOSE Commands.

Mode 3 Command are Block Commands. They Are Routed by the Decoder to the DH Microprocessor for Processing and Validation and Then to the Appropriate RTU for Execution.

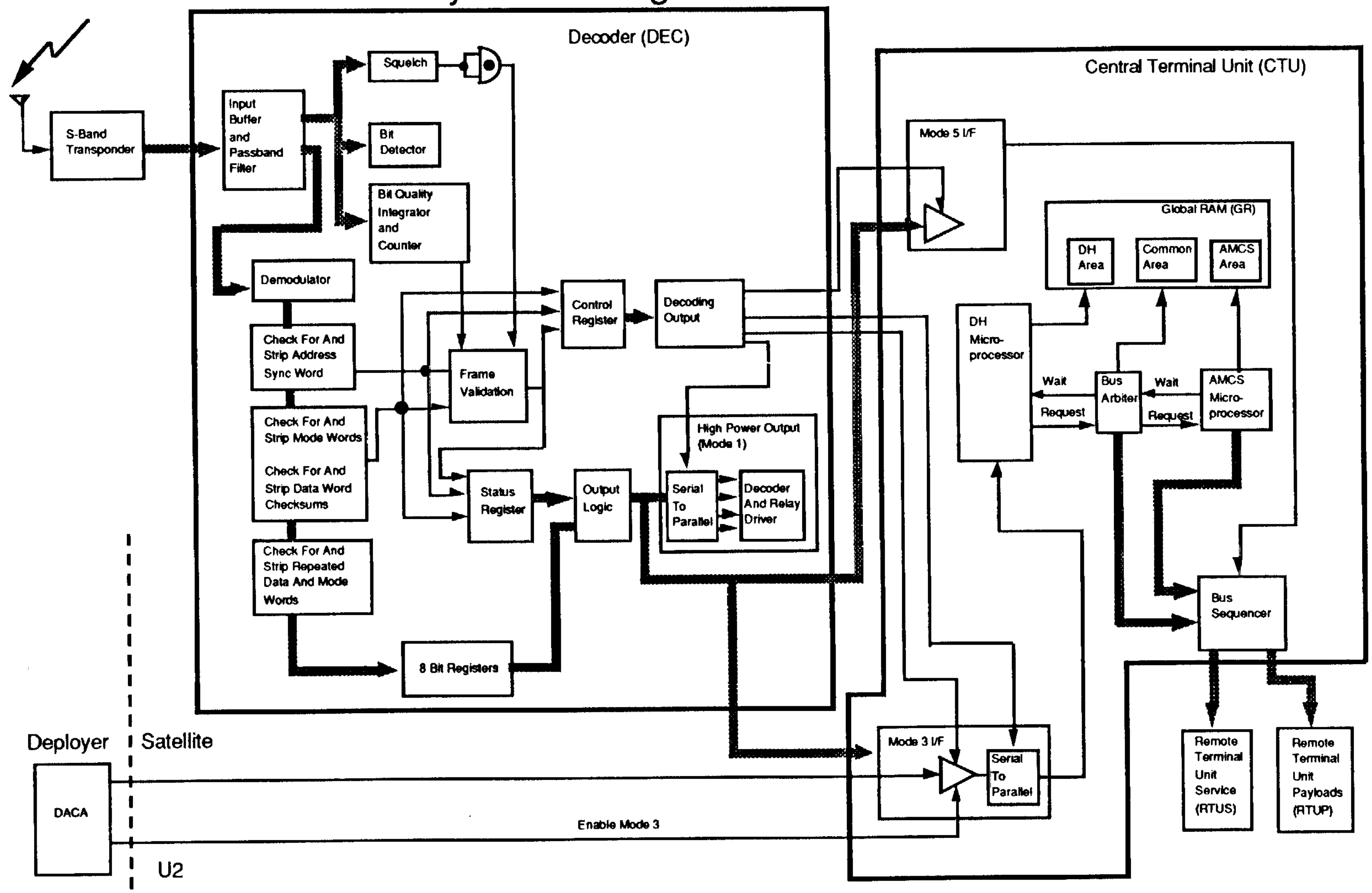
Mode 5 Commands are Routed from the Decoder through the CTU to the Appropriate RTU. These Include ON/OFF or Memory Load Commands.

The Attached Mode Commands are Received through the U2 Umbilical from the DACA. The U2 Interfaces Directly with the CTU. Only Mode 3 Commands May Be Used When the Satellite is Attached. When the RF Link is Established, Mode 1, 3, and 5 Commands May Be Used.

Commands Via DACA U2 and RF Link Cannot Be Sent Simultaneously Because the Data, Clock, and Sampling Line of the U2 Hardline Are Connected to the CTU with the Data, Clock, and Sampling Lines Coming from the Decoder.

On Board Data Handling System

OBDH Command/Telemetry Processing



Telemetry Tracking & Control

Introduction

Telemetry Tracking and Control TT&C System Provides S-Band RF Communications between the TSS Satellite and the Orbiter PI Utilizing a 2-Kbps Uplink and a 16 Kbps Downlink.

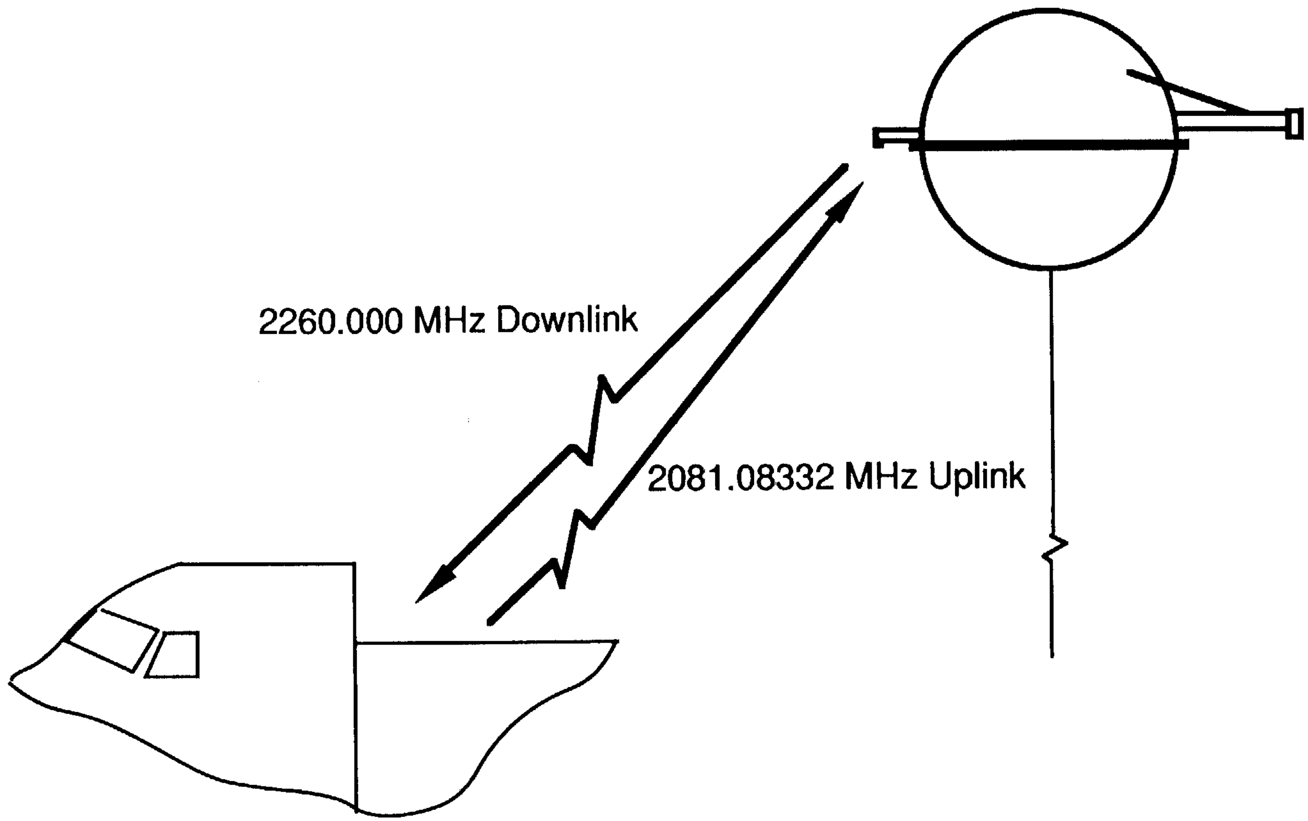
The TT&C is Powered on Prior to Deployer Boom Extension. The S-Band Antenna is Oriented 169° Away from Orbiter PI Before the Boom is Deployed. The Crew Will Attempt to Establish an RF Link with U2 and U1 Connected. If the Link Cannot be Established, it Will Be Established after the Boom is Deployed and the Docking Ring is Rotated to Bring the S-Band Antenna into the PI Field.

The TT&C System Consists of:

- Receiver
- Transmitter
- Diplexer
- S-Band Antenna

Telemetry Tracking & Control

Introduction



Telemetry Tracking & Control

Receiver & Transmitter (Transponder)

The Receiver is a Two-Stage Superheterodyne Device. It is Capable of Maintaining RF Lock over the Orbiter PI Frequency Drifts of ±150 kHz for RF Inputs over the Range of -115 dBm to -30 dBm.

The Orbiter PI Power Levels for Deployment are as Follows:

<80m	Low Power
80m<, ,>250m	Medium Power
>250m	High Power

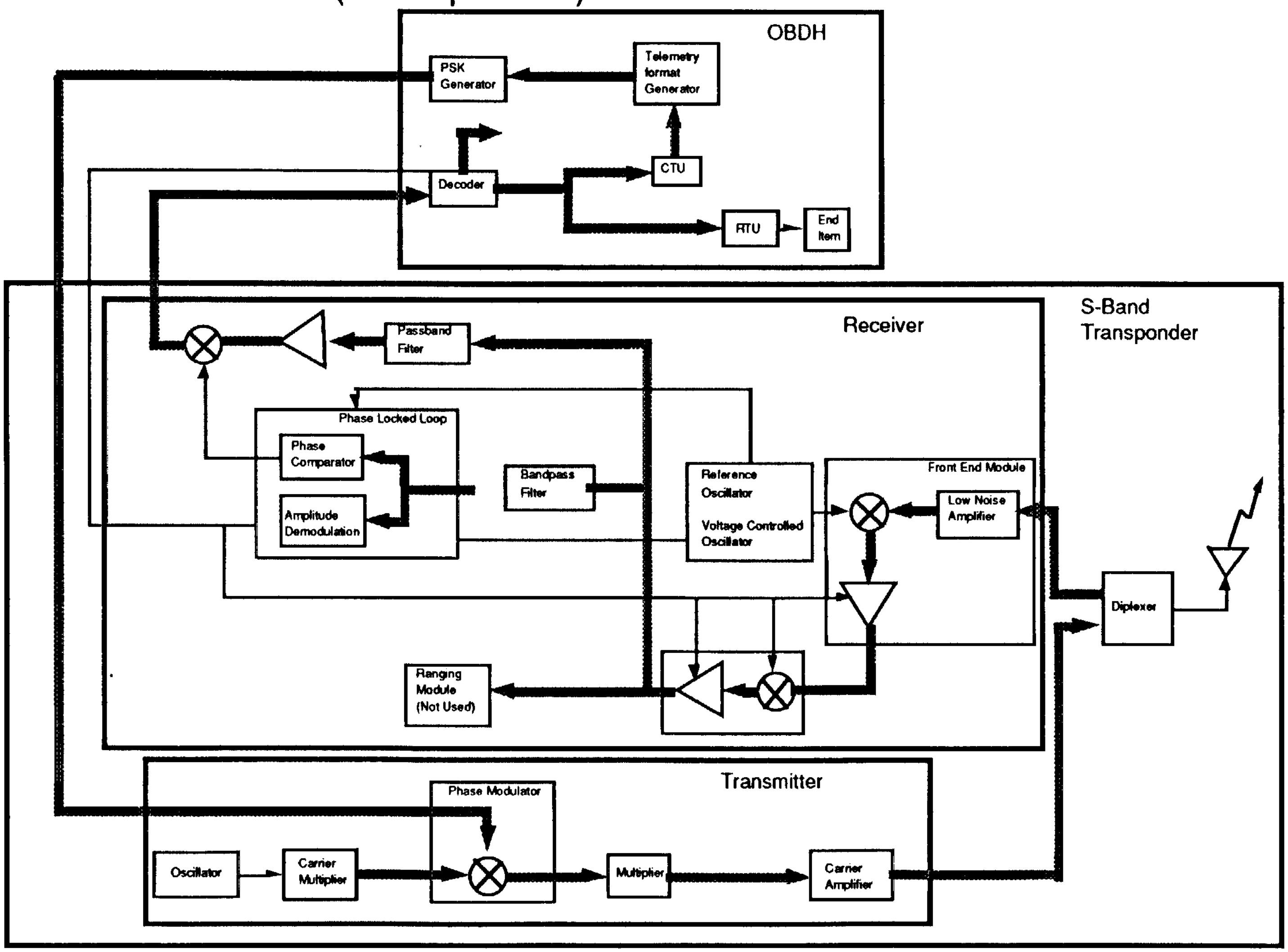
Power Will be Switched from High to Low at 200m during Retrieval

The Transmitter Phase Modulates the TSS Satellite Downlink Data Stream from the OBDH. The OBDH Data Stream is Generated by the Telemetry Format Generator. A 1024 kHz Sinusoidal Telemetry Subcarrier is Binary PSK Modulated by a 16 Kbps (NRZ-L) Data Stream.

The Transmitter Operates in a Noncoherent Mode with a Center Frequency of 2260.000 ±.050 MHz. RF Power Output is 1 W into a 50 ohm Load. The Power Output Level is Not Variable.

Telemetry Tracking & Control

Receiver & Transmitter (Transponder)



Telemetry Tracking & Control

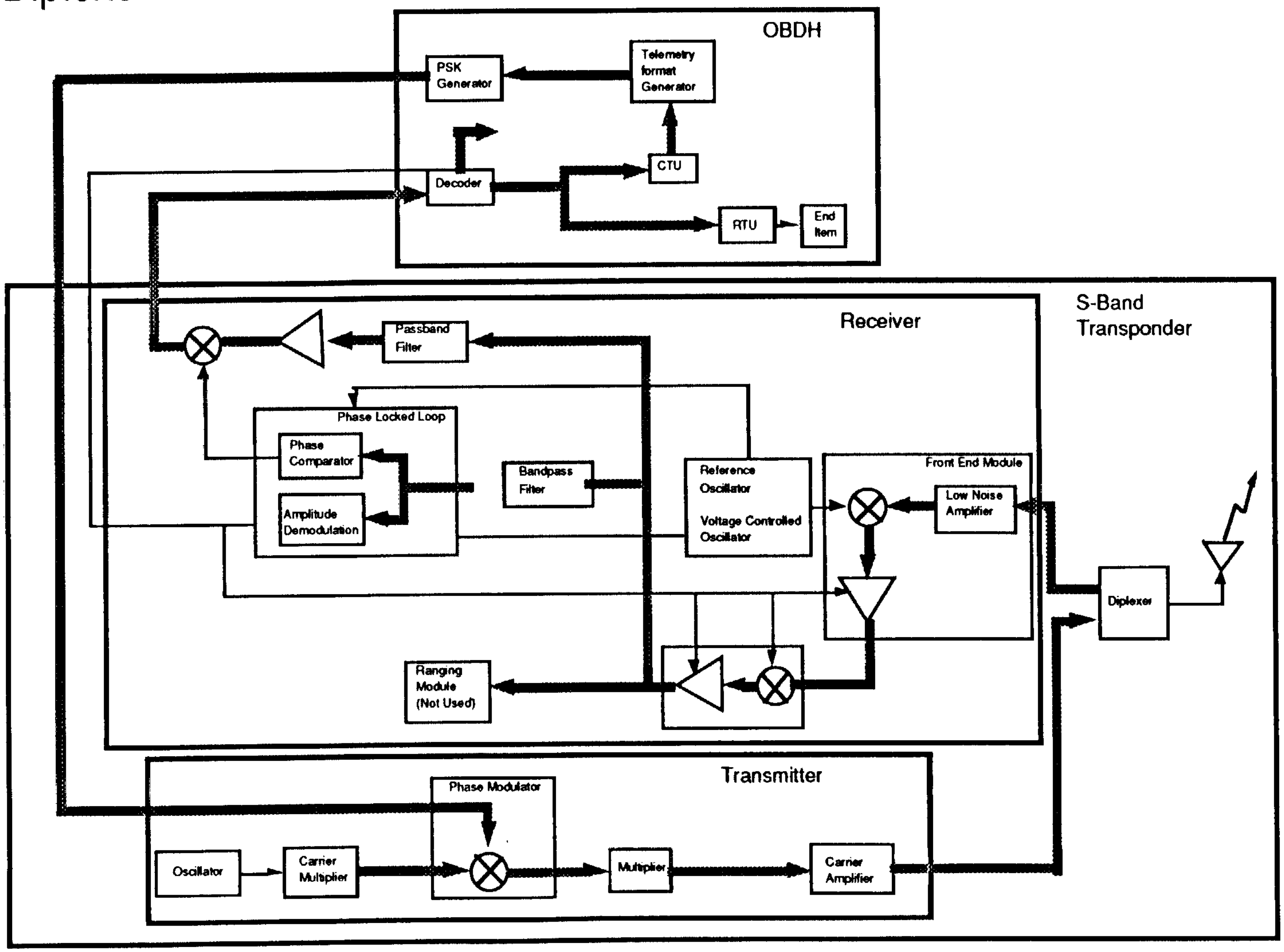
Diplexer

The Diplexer Allows the Transmitter and Receiver to Share a Single S-Band Antenna. An 85 MHz Band-Pass Filter is in the Path to the Receiver.

A 10 MHz Band Stop Filter Centered at the Receiver Frequency is Inserted between the Transmitter and Antenna

Telemetry Tracking & Control

Diplexer



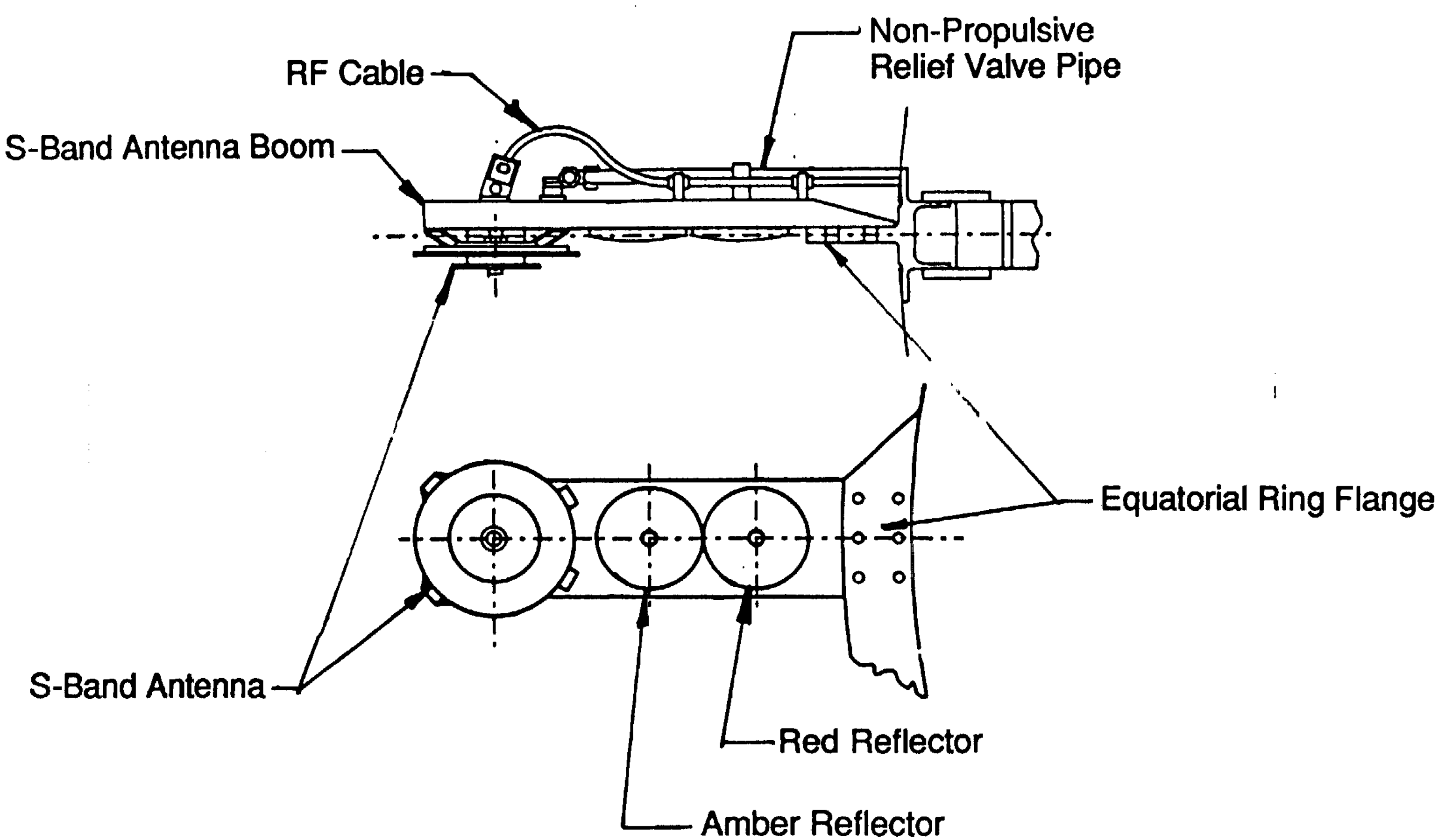
Telemetry Tracking & Control

S-Band Antenna

The S-Band Antenna is a Circular Microstrip Antenna.

It is Located on a Fixed Boom. During nominal Flight, This Boom is Oriented in the Satellite Flight Direction.

S-Band Antenna



S-Band Antenna Boom

Auxiliary Propulsion System

Introduction

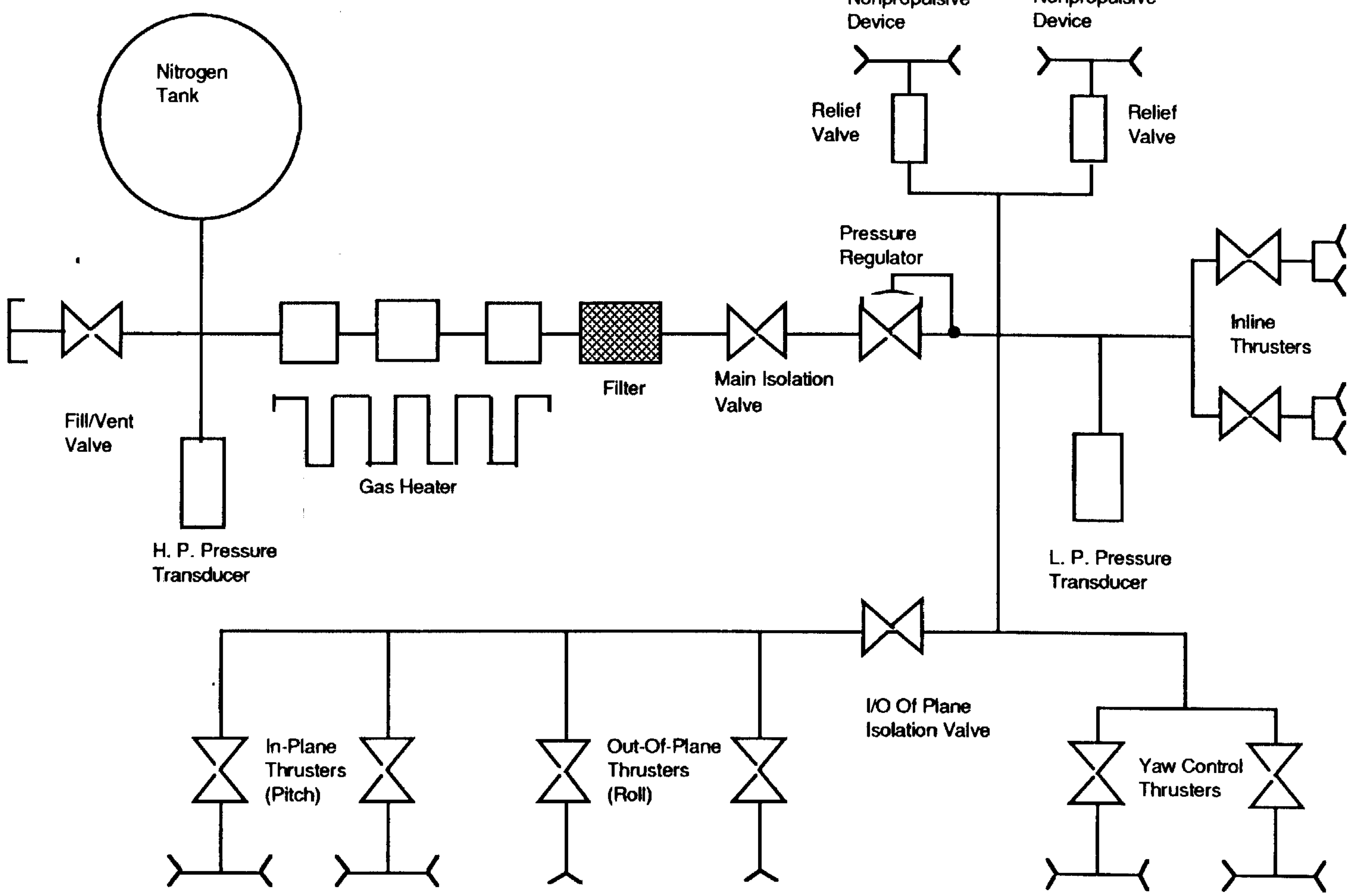
The Satellite Auxiliary Propulsion System (APS) is Used to Supplement the Gravity-Gradient-Induced Tether Tension to Maintain Satellite Attitude and Control during Flight.

The APS Consists of:

- 1 Propellant Tank
- 1 Pressure Regulator
- 1 In-Line Filter
- 3 Heaters
- 2 Isolation Valves
- 1 Fill/Vent Valve
- 2 Relief Valves
- 2 In-Line Thrusters
- 2 Yaw Thrusters
- 2 In-Plane Thrusters (Pitch)
- 2 Out-of-Plane Thrusters (Roll)

Auxiliary Propulsion System

Introduction



APS Schematic

Auxiliary Propulsion System

Tank

The Tank is a Spherical, Welded Titanium Vessel with a of Volume 0.22 m^3.

The Tank Mounts to Equatorial Floor on 16 Feet.

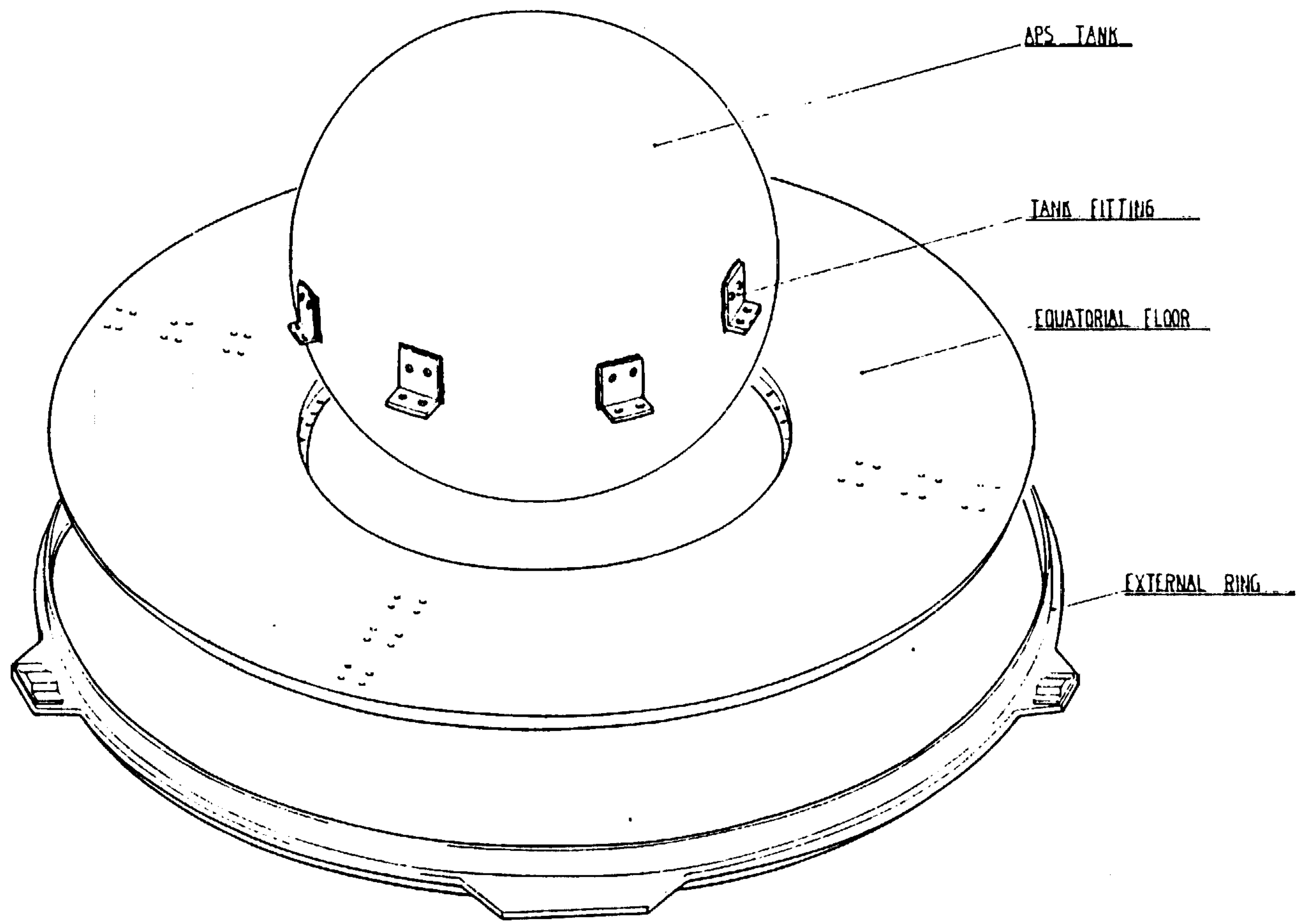
Pressure Capability of the Tank is:

300 Bars	Operating
600 Bars	Burst

Capacity (Pre-Mission):

60.18 Kg GN_2 at 50 C

Auxiliary Propulsion System
Tank



Auxiliary Propulsion System

Pressure Regulator, In-Line Filter, & Heaters

The Pressure Regulator is Located Immediately after Main Isolation Valve. It is Designed to Hold Outlet Pressure at 153.4 to 142 psi.

A Flow Restricting Snubber Valve with a Restrictor Orifice is an Integral Part of the Regulator on the High Pressure Side. It is Designed to Restrict Flow if the Regulator and Snubber Fails and the Relief Valves Open.

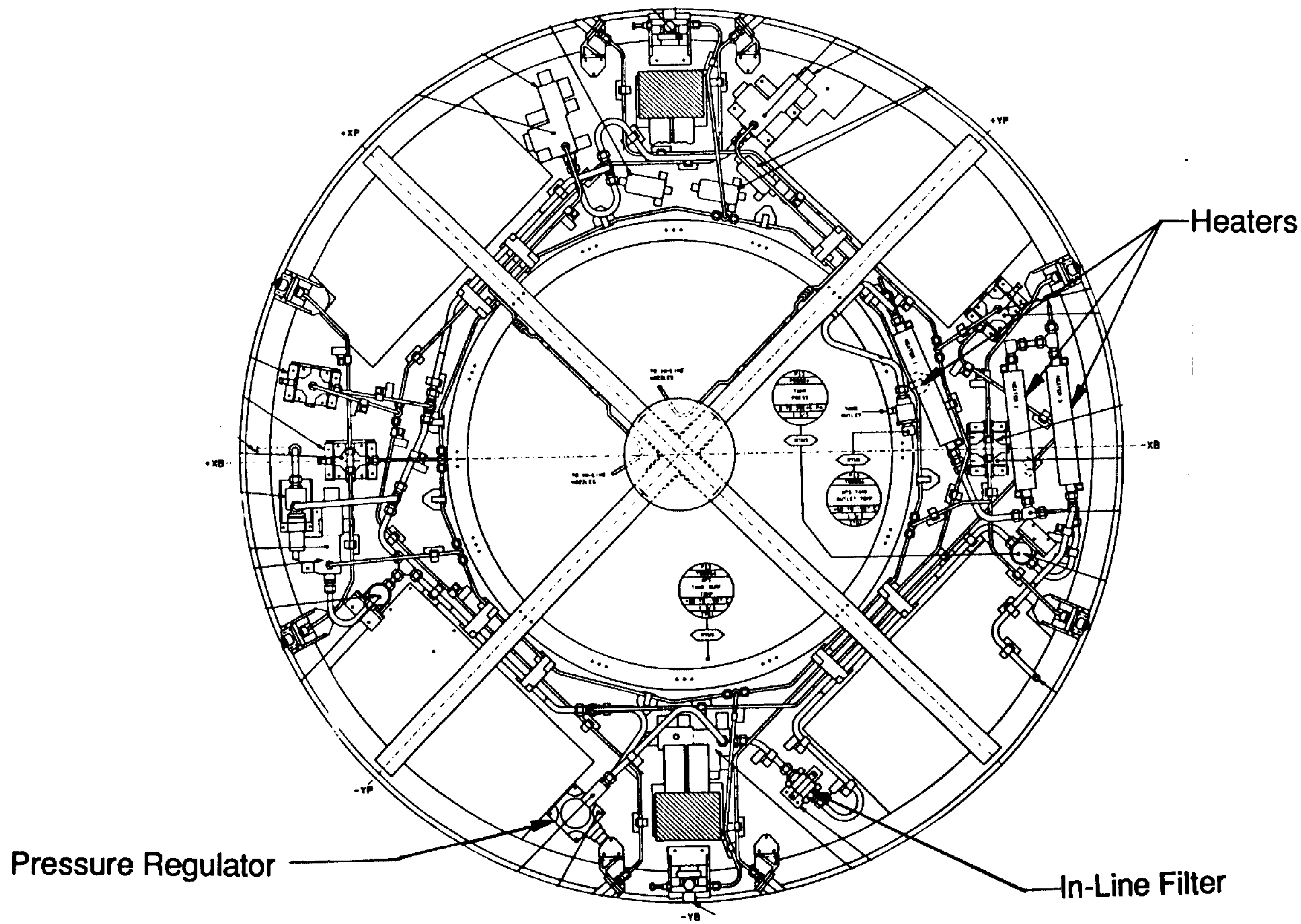
The In-Line Filter is a Stainless Steel High-Pressure Propellant Filter. It Filters Particles That Are 10 Microns Absolute.

Three Heaters are Located on the Pipe between the Tank and the Main Isolation Valve to Warm the GN_2 for Additional Impulse and to Hold Component Temperatures above Their Critical Limits. The Heaters Are Always on When the In-Line Thrusters Are on.

The System Consists of Three Heaters and Three Controllers. The Controller Set Points Are -34 C, -32 C, and -30 C. The Heaters are Nickel-Chromium-Iron Wire Insulated with MgO. They Are Wrapped Around the Pipe and Consume Approximately 7W.

Auxiliary Propulsion System

Pressure Regulator, In-Line Filter, & Heaters



Auxiliary Propulsion System

Isolation, Fill/Vent & Relief Valves

The Satellite APS Has 2 Stainless Steel Latching Solenoid Isolation Valves. These Isolation Valves Isolate Some or All of the Thrusters from the Tank. The Main Isolation Valve Closes Whenever the APS is Disabled. The In/Out of (I/O) Plane Isolation Valve Latches Open or Closed.

Telemetry That is Available Includes Body Temperature and Open/Closed Status.

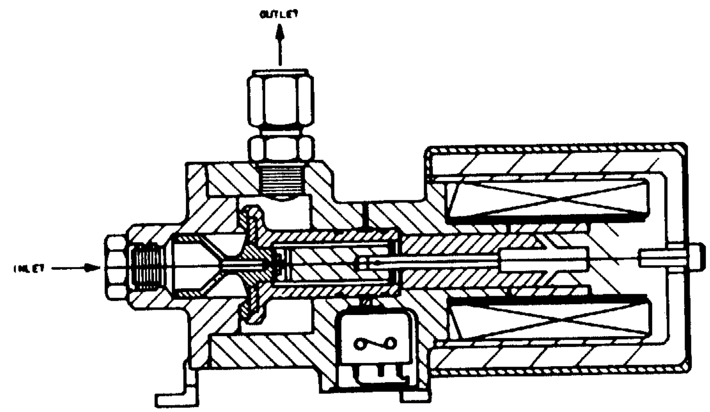
The Fill/Vent Valve is a Stainless Steel, Manually Operated Valve for Direct Tank Loading and Unloading by Ground Support Equipment. The Valve Is Capped and Sealed for the Mission.

The APS Relief Valves Prevent Over Pressurization of the Low Pressure Components in the Event of a Pressure Regulator Failure.

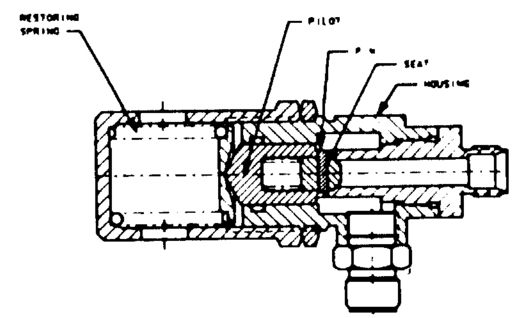
A Spring Loaded Seat in an Aluminum Body Displaces at 260 $\pm$13 psia, and Reseats at 220 psia. The Outlet Pipe Runs the Length of the S-Band Antenna Boom and Branches at a Tee into Two Nonpropulsive Vents.

Auxiliary Propulsion System

Isolation & Relief Valves



Isolation Valves



Relief Valves

Auxiliary Propulsion System

In-Line & Yaw Thrusters

The Satellite APS Has 2 Sets of In-Line Thrusters. For the TSS-1 Mission, the In-Line 2 Will Be Used. Each Set Consists of a Latching Solenoid Valve and Two Nozzles.

The In-Line 2 Set Has 1.792 to 1.943 N of Thrust.

Telemetry That is Available Includes Temperature and Open/Closed Status.

The APS Also Has 2 Nonlatching Yaw Thrusters, Each of Which Consist of an Aluminum Body with 2 Nozzles Located 180° Apart

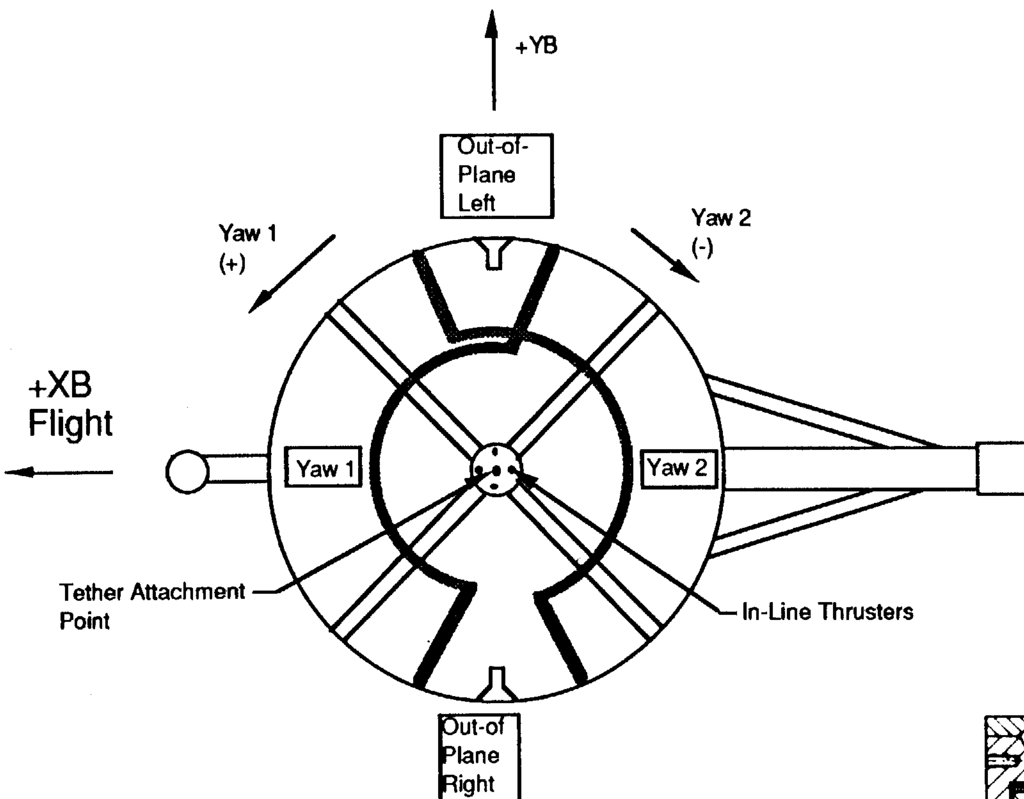
Yaw 1 Produces Counterclockwise Rotation, and Yaw 2 Produces Clockwise Rotation When Looking from the Tether Attachment Point.

The Yaw Thrusters Produce 0.983 to 1.139 Nm of Torque. They are Fired by AMCS Software Control.

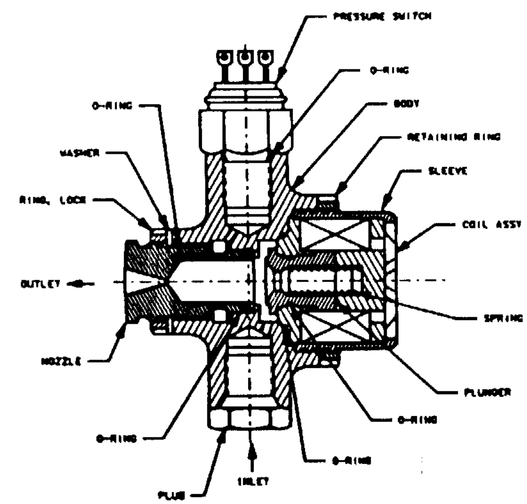
Telemetry Available Includes Yaw 1 Valve Body Temperature and Both Valve Open/Closed Statuses Via a Pressure Microswitch. The Pressure Microswitch Registers Open When the Valve Outlet Pressure is Greater Than 100 psia and it Registers Closed When the Valve Outlet Pressure is Less Than 80 psia.

Auxiliary Propulsion System

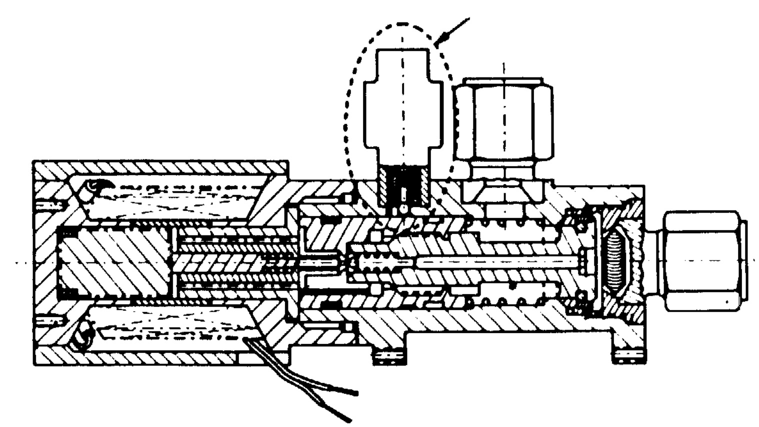
In-Line & Yaw Thrusters



Yaw Thruster Directions



Yaw Thrusters



In-Line Thrusters

Auxiliary Propulsion System

In-Plane & Out-of-Plane Thrusters

The APS Has 4 In-Plane and Out-of-Plane (I/O Plane) Thrusters. These Thrusters Consist of Aluminum Nonlatching Solenoid Valves and Nozzles. The I/O Plane Thrusters Are Aligned with the Satellite Body Axes.

The Thrusters Provide 2.646 to 3.066 N of Thrust. The 4 In-Plane Nozzles Are Canted 30° Toward -Z to Provide Some Pitch Control of the Satellite. The 4 Out-of-Plane Nozzles Are Canted 20° Toward -Z to Provide Roll Control.

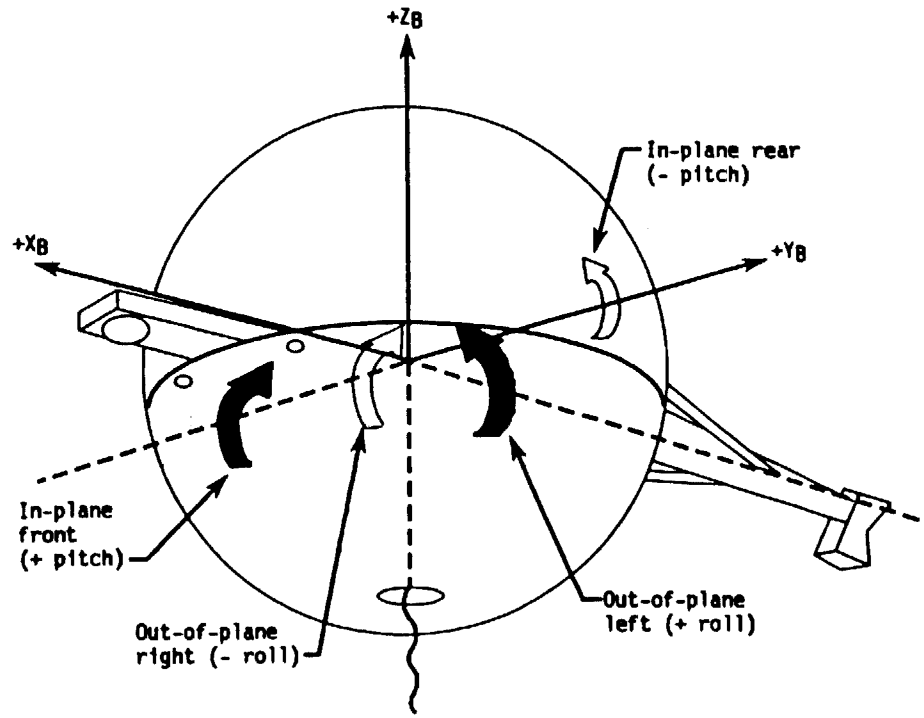
The Crew and the MCC Can Control the Satellite I/O Plane Angles. The Thrusters are Commanded Manually and Are Cycled Closed by Software, Based on a Commanded Firing Duration. A Manual Software Override Feature is Available to Close a Valve.

The I/O Plane Thrusters Are Disabled by Closing the I/O Plane Isolation Valve When Tether Deployed Length is <200 m.

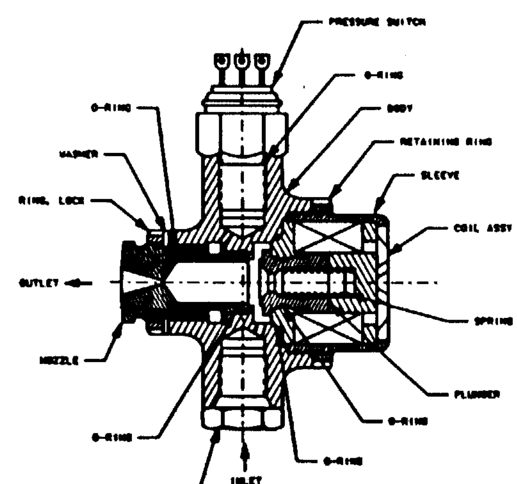
Telemetry Available Includes Valve Open/Closed Status Via Pressure Switch and Valve Body Temperature Except for the Out-of-Plane Right Thruster.

Auxiliary Propulsion System

In-Plane & Out-of-Plane Thrusters



Note: In-plane nozzles canted 30°.
Out-of-plane nozzles canted 20°.



In-Plane Thrusters (Pitch)
Out-Of-Plane Thrusters (Roll)

Attitude Measurement & Control System

Introduction

The Attitude Measurement and Control System (AMCS) is Responsible for Performing Three-Axis Satellite Attitude Determination and Maintaining Yaw Attitude and Rate Control.

The Satellite Pitch and Roll Axes are Naturally Stabilized by Gravity Control Gradient Torques. the Satellite is Equipped with I/O Plane Thrusters Which Have Been Canted to Allow Manual Control of Pitch and Roll Angles for Damping of Tether Skip Rope.

All AMCS Command Telemetry Processing is Performed by the AMCS Microprocessor. The Software Calculates Satellite Attitude from the Gyro Outputs. It Also Uses the Earth Sensor Data to Continuously Correct the Estimates of the Rate Being Integrated When the Procedure is Enabled by Ground Command. Sun Sensor Measurements Are Not Used for on Board Attitude Determination.

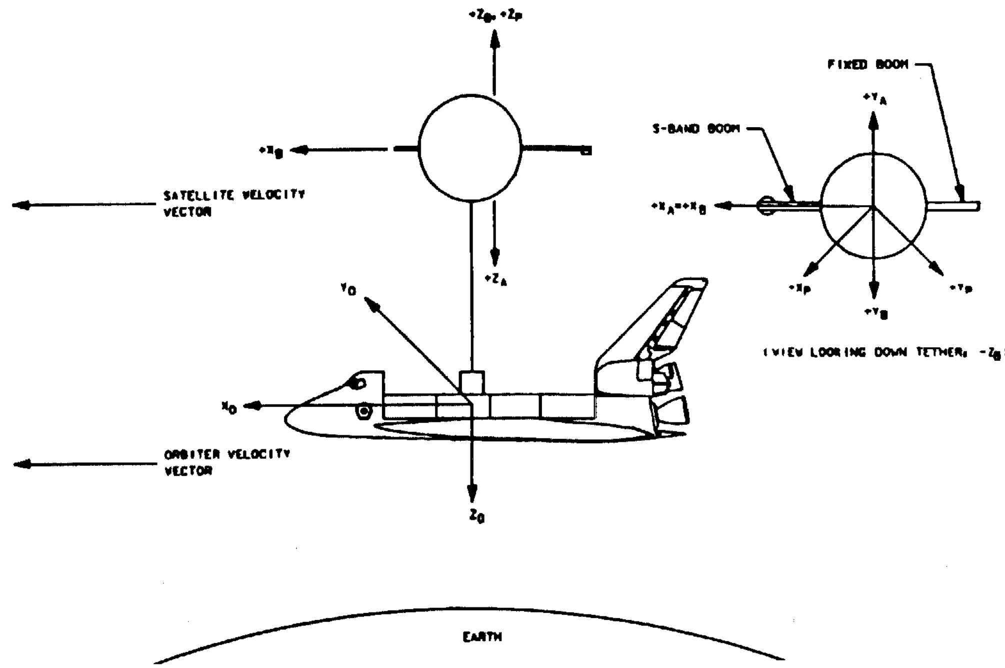
The AMCS Has 2 Operational Modes, Coarse or Fine. OBDH Software Automatically Changes the Gyro Mode Based on the Gyro Rate. The Mode Can Also be Changed by MCC Command.

The Satellite Software Uses Earth Sensor Data to Offset the Effects of Gyro Random Drift Errors. Constant Drift Will Be Measuring during Predeployment Gyro Calibration. Constant Drift Values Will Be Loaded into the OBDH Global RAM and Used by AMCS Software to Correct the Gyro Incremental Angles.

The AMCS is required to Set and Maintain a Spin Rate of 1 ± 0.1 rpm, to Maintain Yaw Angles with a Maximum Error of $\pm 3°$, and to Determine Satellite Attitude about All Three Axes with an Accuracy of $\pm 1°$.

Attitude Measurement & Control System

Introduction



(X_O, Y_O, Z_O) - ORBITER ATTITUDE REFERENCE (LVLH)

(X_B, Y_B, Z_B) - SATELLITE BODY REFERENCE FRAME [1]

(X_P, Y_P, Z_P) - SATELLITE STRUCTURAL REFERENCE FRAME (USED FOR MECHANICAL INTEGRATION) [2]

(X_A, Y_A, Z_A) - AMCS BODY COORDINATE SYSTEM (ROLL, PITCH, YAW AXES) [3]

(X_{RS}, Y_{RS}, Z_{RS}) - SATELLITE ATTITUDE REFERENCE (LVLH) [3]

REFER TO DWGS TBS TO SEE AMCS SENSOR MOUNTING CONFIGS RELATIVE TO AXES

[3] ORIGIN IS AT SATELLITE CENTER OF MASS

[2] THE ORIGIN OF THIS FRAME IS IN THE GEOMETRIC CENTER OF THE SATELLITE WHICH IS 75 MM ABOVE THE EQUATORIAL PLANE

NOTES: [1] ORIGIN IS AT GEOMETRIC CENTER OF SATELLITE

Attitude Measurement & Control System

Data & Command Interfaces

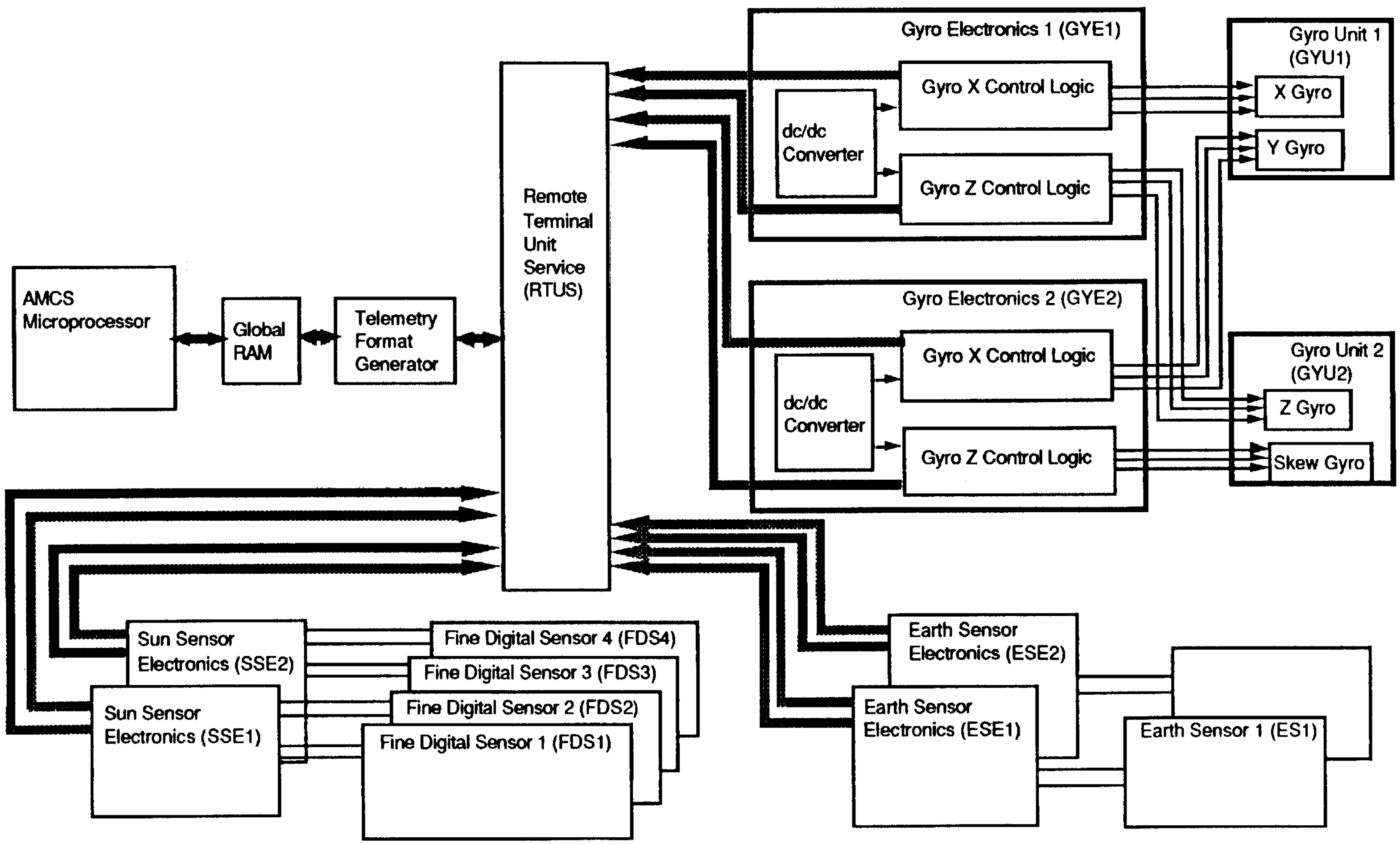
Each Gyro Provides 1 16 bit Serial Word to the RTUS at a Frequency of 8 Hz, or 2 Words Per Gyro Per Satellite Minor Frame Every 4 Minor Frames. This Data is Called Gyro Incremental Angle A and Gyro Incremental Angle B.

The Data Word Contains Status Bits for the Gyro dc/dc Convertor, Wheel, Heater, and Gyro Mode. The Remaining 123 bits Are the Complement of the Cumulative Value over 1 OBDH RTC of the Torquing Current Counting Register.

The Gyro Outputs Are Accumulated by the Software over 6 OBDH RTCs. The Outputs Are Divided by Time, Scaled, and Placed into Telemetry As an Analog Measure of Gyro Rate. Gyro Rate is an Average of the Rates Experienced over Six Consecutive RTCs.

Attitude Measurement & Control System

Data & Command Interfaces



Attitude Measurement & Control System

Gyroscopes

The AMCS Uses 4 Single-Axis Rate Integrating Gyroscopes (RIGs) to Sense the Angular Variations Around the Satellite's 3 Principal Body Axes.

The System is Grouped into 2 Gyrometer Units (GYUs) GYU1 and GYU2. GYU1 Contains the X and Y Sensing Gyros Which Are Mounted on the $+Y_p$ Meridional Panel. GYU2 Contains the Z Sensing Gyro and a Gyro Skewed 73° between the X and Y Axes. It is Mounted on the -Yp Panel. The Skew Gyro Provides Increased Yaw Rate Sensing Capability and Redundant Roll Measurements.

GYU1 and GYU2 Receive Power from 2 Gyro Electronics Packages (GYE), GYE1 and GYE2. Each GYE Power Subassembly is Enabled by MCC or Software Issued Commands. Power is Disabled by MCC Command.

When a GYE Power-On Command is Received, the GYE Automatically Powers on the GYU Heaters Which Maintain a 71.5 $\pm$2 C Set Point. The Drive Wheel is Enabled When Temperature Reaches 55 $\pm$2 C. The Wheel is Disabled if the Temperature Drops Below 55 C. If the Gyro Temperature Exceeds 85 C, the GYE Disables the Thermal Control and the Wheel for That Gyro. During Post-Retrieval, the Deployer Hot Nest Heaters Maintain the Gyros above -10°C.

Attitude Measurement & Control System

Gyroscope Characteristics

Input rate limits (fine for performance) - X, Y, Z, and skew gyros	0 to 0.8 deg/sec (0 to 0.1333 rpm)
Input rate limits (coarse for performance) - X, Y, Z, and skew gyros	0 to 2.0 deg/sec (0 to 0.333 rpm)
Incremental angle value at saturation (fine) - X, Y, Z gyros - Skew Incremental angle value at saturation (coarse) - X, Y, Z gyros - Skew	+ 2047 x <SB ≈ 1 deg/sec 1638 x <SB ≈ 2 deg/sec
Estimated constant drift - Fine mode - Coarse mode	Spec valve / Actual ± 1 deg/hr maximum / -0.3646 to 0.922 ± 2.5 deg/hr maximum / -0.735 to 0.708
Estimated random drift - Short term - Long term	≤.02 deg/hr ≤.015 deg/hr
16-bit SD word - At power on - After undervoltage recovery	B0: dc/dc conv stat = ON B1: Heater stat = ON B2: Wheel stat = Not running (running at temp > 55° C) B3: Mode = Last mode selected B4: ⋮ }Gyro incremental angle (depending on input rate) B15: B0: dc/dc conv stat = OFF then ON after 2 gyro acquisitions B1: Heater stat = OFF then ON after 2 gyro acquisitions B2: Wheel stat = Anything then running after 2 acquisitions B3: Mode = Anything then running after 2 acquisitions B4: ⋮ }Gyro incremental angle = 0 then meaningful after 2 acquisitions B15:

Attitude Measurement & Control System

Earth Sensors

The Earth Sensor (ES) Assembly is Designed to Provide Relatively Accurate Pitch and Roll Body-Referenced Attitude Data during the Orbit, Unless the Field-of-View (FOV) is Momentarily Blocked. A FOV Obstruction is Avoided by Disabling the Software Processing of ES Data Whenever a Blockage is Expected. The OBDH Software Uses ES Data to Correct the Satellite Attitude Values Computed from the Gyro Data. The ES Assembly Can Detect a ±45° Pitch Angle and ±27° Roll Angle When the Satellite is in Hold. When the Satellite is Spinning, Pitch and Roll Continuously Change and ES Angles Are Not Used.

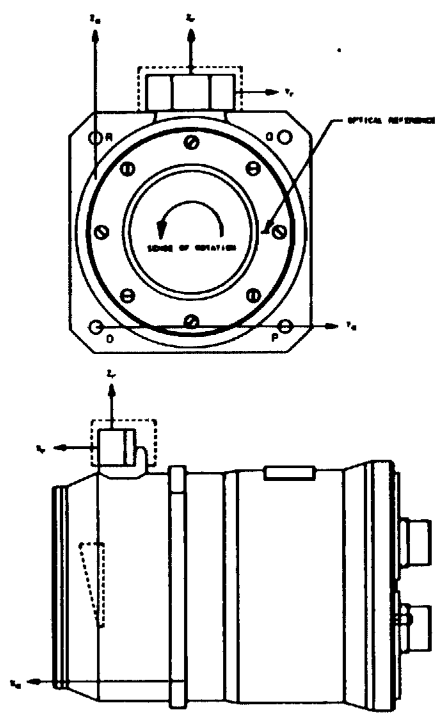
The ES Assembly Consists of Two Optical Heads, ES1 and ES2, Pointing along the Satellite $+Y_b$ and $-Y_b$ Axes. Each ES Head Has ES Electronics (ESE) to Provide Signal Conditioning and Power.

An Offset Motor Driven Prism Rotates to Scan a Bolometer through a 45° Sweep. Each ESE Then Generates Two Chord Angles. Chord 1 is Proportional to the Scan Angle between the Space-to-Earth Crossing and a Zero Reference. Chord 2 is Proportional to the Scan Angle between the Earth-to-Space Crossing and a Zero Reference. The Chord Angles Are in Units of Motor Encoder Steps. 1 Step Equals 0.01° of Shaft Rotation. Each ES Head Generates a Chord 1 and a Chord 2. The OBDH Software Calculates a Chord 1 Mean and a Chord 2 Mean Which Are Then Processed in Attitude Computation Software

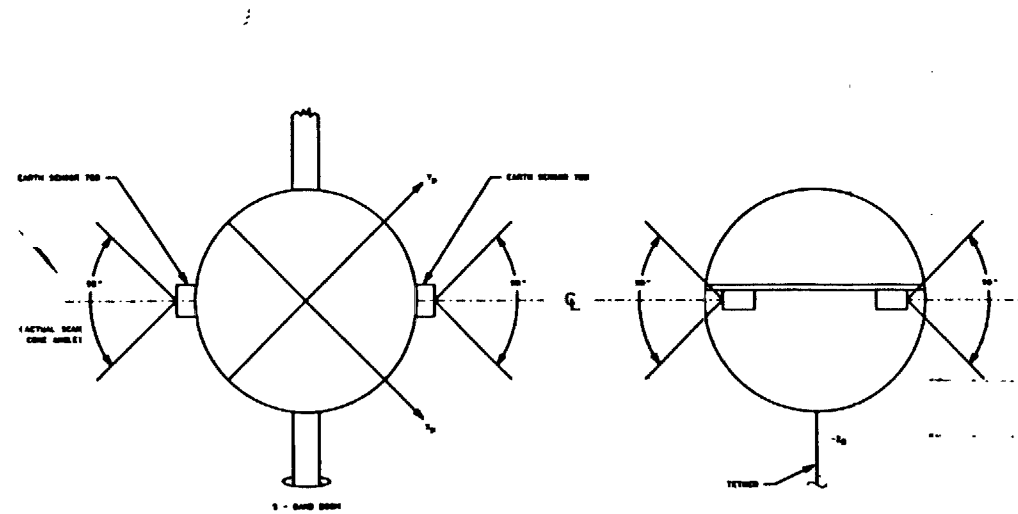
Each ES Sends 2 16-bit Serial Digital Words to the OBDH RTUS Corresponding to Chord 1 and Chord 2. Each ESE Provides an Analog Measurement of Head Temperature and an Analog Voltage Indicating Sun Presence. If the Voltage is Greater Than 4.7 Volts, the Sun is in the ES Field Of View (FOV). No Damage Occurs if the Sun is in the ES FOV. A Discrete Status Will Indicate Fault If No Space-to-Earth or Earth-to-Space Crossing Has Occurred, or If the Bolometer Motor Speed is > 106% of Expected, or If the ESE Power Supply Voltage is Low. If the ESE Status Bit is FAULT, Software Processing of ES Data is Inhibited.

Attitude Measurement & Control System

Earth Sensors



EARTH SENSOR
(TYPICAL)



VIEW LOOKING DOWN TETHER (-Z_B)

EARTH SENSORS FIELD OF VIEW

Attitude Measurement & Control System

Sun Sensors

The Sun Sensors Are Used to Measure Satellite Attitude with Respect to the Sun. The Telemetry is Not Used by the OBDH for Attitude Computation, But is Included in the Satellite Downlink for Use in Ground Computations during Flight and Post Flight.

The Sun Sensor Assembly is Composed of 4 Fine Digital Sensors (FDSs). Each is Composed of 2 Identical Sensing Head Units with a Photoengraved Slit, Behind Which is a Linear Array of Photo Detectors. The Slit Length Defines an Off-Axis FOV of ±45° to the Centerline. The Sensing Head Units Are Mounted So that the Slits Form a Total FOV of 90° X 90°. Each FDS is Mounted So That the FDS Centerline is 11° from the Meridional Panel.

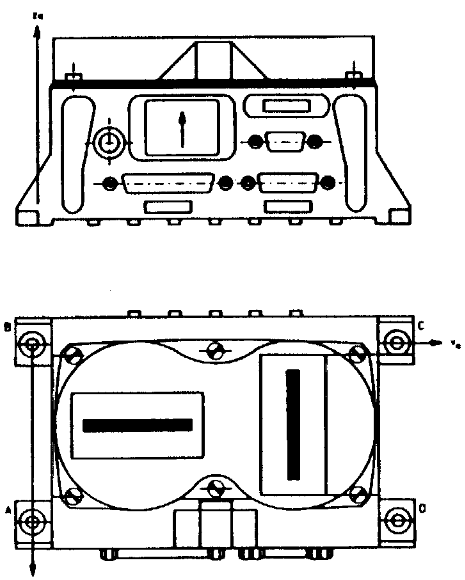
The FDS Photo Detector Pixels are Continually Scanned for Voltage by Digital Summation. By Averaging of the Number of Pixels Registering a Voltage, the Apparent Sun Image Center is Determined in Terms of 1/8 Pixel Steps from the End of the Sensing Unit Head Slit. The Number of Pixel Steps is Converted into an Angle, Alpha Which Equals Yaw, and Beta Which Equals Pitch/Roll.

Each Sun Sensor Assembly Has a Corresponding Sun Sensor Electronics (SSE) which Receives Raw Angle Data from Each of Two FDS Units. The SSE Chooses the FDS That Has the Sun in the FOV and Calculates the Alpha and Beta Angles. If Both FDS Have the Sun in the FOV, the SSE Selects One Head Based on Internal Priorities.

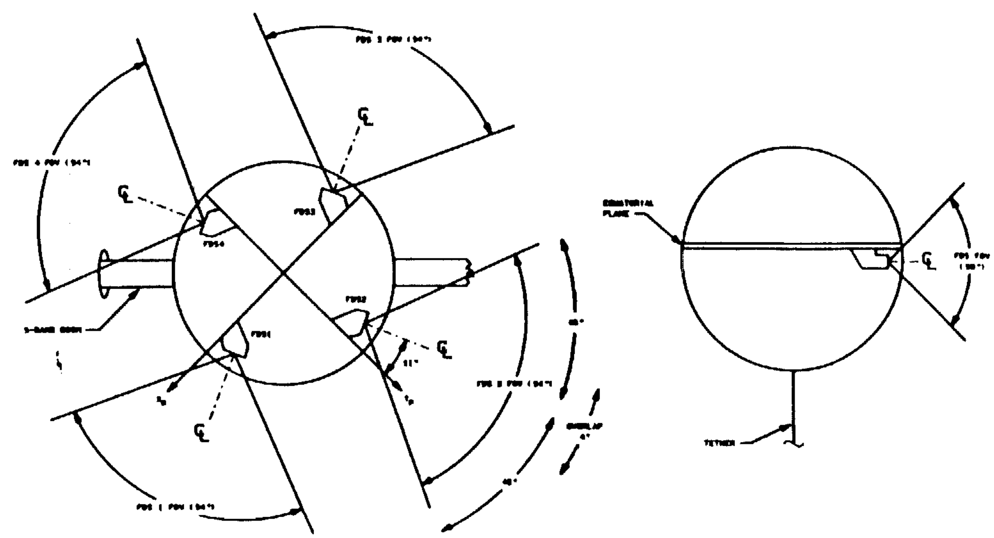
Alpha and Beta are Provided of the OBDH RTUS as 2 16-bit Serial Digital Words. The Last 2 Bits in the 16 -bit Word Indicate Which FDS Was Selected and if the Data Are Valid. Data Not Valid if the Sunspot Signal is Not over a Preselected Threshold

Attitude Measurement & Control System

Sun Sensors



SUN SENSOR (TYPICAL)



VIEW LOOKING DOWN TETHER (-Z$_B$)

SIDE FIELD OF VIEW (TYPICAL)

SUN SENSORS FIELD OF VIEW

Thermal Control System

Passive Elements

The Thermal Control System (TCS) Uses Passive and Active Thermal Elements to Control Satellite Temperatures Within Acceptable Operating Ranges. Passive Control includes External and Internal Paint, Multilayer Insulation (MLI) Blankets, Analog Temperature Sensing, and Discrete Temperature Monitoring. Active Control Consists of Several Heaters in the Satellite Gyroscope Heaters and Hot Nest Heaters in the Deployer's Satellite Support Structure for Predeploy and Postretrieval Control.

The External Surface of the Satellite is Painted with White Conductive NASA GSFC NS43-C. The Markings on the Exterior are Painted with Electrodag 501 Black Paint. The Satellite Equatorial Ring is Unpainted, with a Alodine Finish. The Internal Cavities and Components are Painted Black.

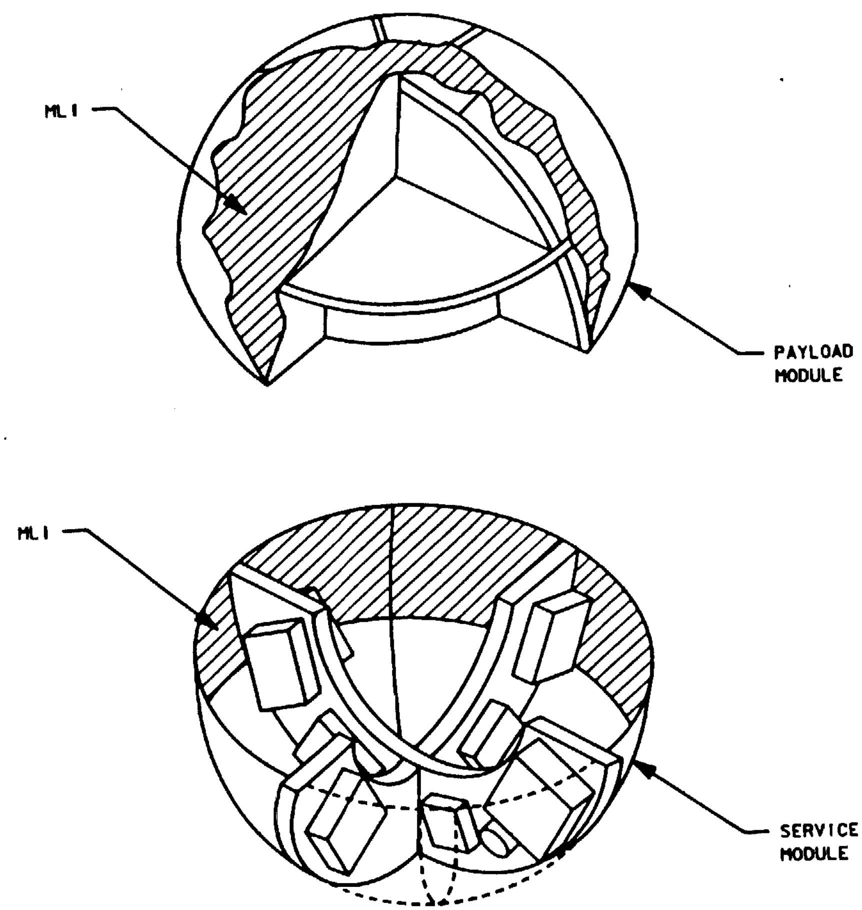
The Internal Surface of the Satellite Skin is Covered with MLI. The Bottom Half of the Service Module Skin is Not Covered with MLI to Provide a"Radiator" Section. This Allows Heat Transfer from the Deployer Hot Nest Heaters during Post-Retrieval to Warm the Gyros. It is Also the Only Means of Heat Expulsion when the Satellite is Deployed. The EDY Boom is Covered with a Blanket of Teflon-Coated Fiberglass Cloth. The Holes in the Skin for the Sun and Earth Sensors are Closed with Aluminum Shields which are Painted on the External Surface. The Internal Surfaces are Covered with MLI.

The TCS Uses 24 Thermistors to Sense Internal Satellite Structure Temperatures. The Thermistor's Accuracy is ±4 C. 32 Platinum Resistances are Used to Give 16 Skin Temperatures. Accuracy is ±10 C

5 Thermostats, Which are are Powered by a Discrete Output Low in the PF1 MDM, are Utilized to Provide Discrete Temperature Monitors, to Determine Selected System Component Temperatures Before the Satellite is Powered. These Discretes are BAT 1/2 TEMP UPPER EXTREME, BAT 3/4 TEMP UPPER EXTREME, GYRO TEMP LOWER EXTREME, PM TEMP UPPER EXTREME, SM TEMP UPPER EXTREME.

Thermal Control System

Passive Elements



Thermal Control System

Active Elements

Active Elements in the TCS Include the Satellite Heaters. There are 8 Heaters on the Payload Module Central Cone. These Heaters are Powered by the Pallet PCB Via U1 and are Used Only during Predeployment Checkout.

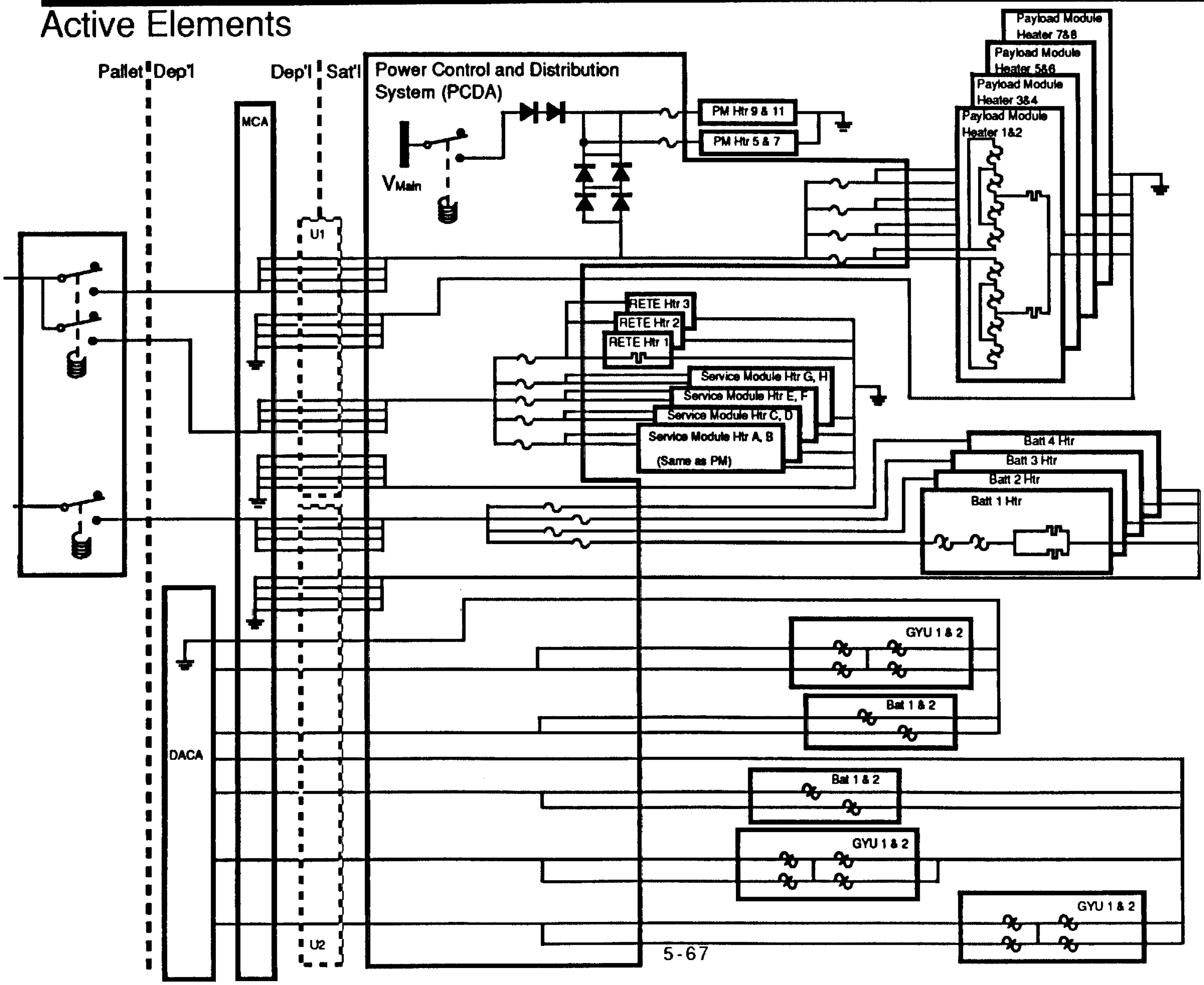
There are 4 Heaters on the Payload Module Polar Panels Which are Powered from the U1 or the Satellite Main Bus. These Heaters Can be Used after U1 Separation.

There are 8 Heaters on the Service Module Meridional Panels Which are Powered through U1. These Heaters are Used Only during Predeployment Checkout.

Equipment Heaters Include 2 Redundant, Thermostatically Controlled Heaters per Battery. The Gyroscopes are Thermally Controlled by Heaters Until the Postretrieval Phase. The Gyros are Then Heated by Hot Nest Heaters after the Satellite is Latched Down.

Thermal Control System

Active Elements



Deployable Retrievable Booms

Overview

The Satellite Has 2 Deployable Retrievable Booms (DRB) Which are Used to Support the Research on Electrodynamic Tether Effects (RETE) Experiment Sensors. The Booms Allow the Sensors to be Inserted into an Environment up to a Distance of 2.41 m (7.9 ft) from the Satellite Skin.

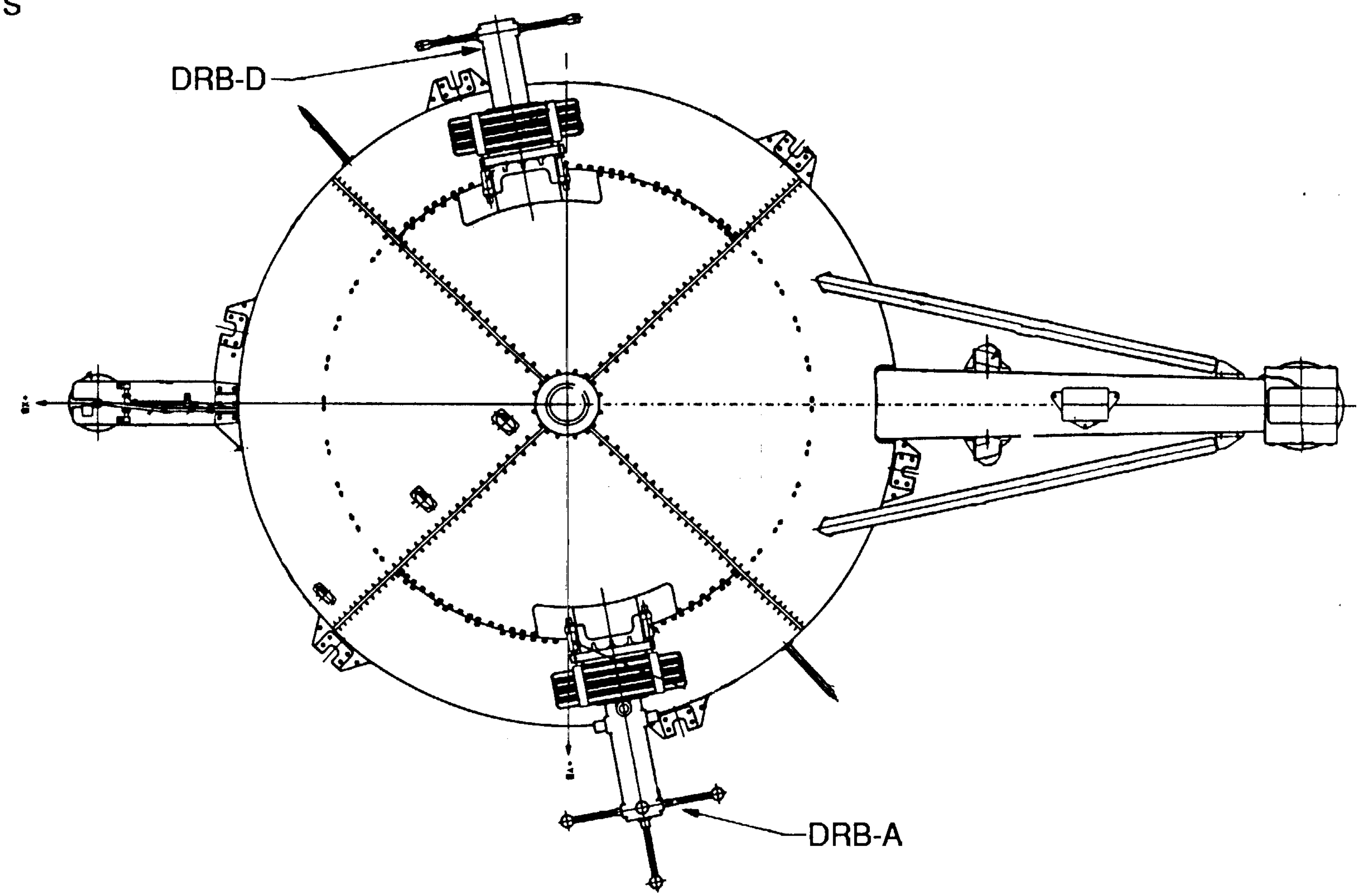
The DRB System is Composed of 1 Electronics Unit (DRBE) That Drives 2 Identical Boom Mechanical Units (DRBA & DRBD). The RETE-AC Experiment is Located on DRB-a. The RETE-DC Experiment is Located on DRB-D. The DRBs are Mounted on the Satellite Payload Module Upper Platform, 180° Apart.

The DRBs Will be Deployed On-Station 1. They are Retrieved and Latched Prior to Docking. The DRBs Can be Spring-Jettisoned in Case of Failure to Relatch in Order to Clear the Orbiter Cargo Bay Envelope.

The Total Mass of Each DRB is 6.3 Kg. The Jettisonable Mass is 5 Kg.

Deployable Retrievable Booms

Locations



Satellite from +Z

Deployable Retrievable Booms

Structure

The Deployable Portion of the DRB Structure is Composed of 8 Concentric Tubes of 6061-T6 Aluminum. 1 of the 8 Tubes is Fixed and the Other 7 Tubes Telescope. The Fixed Tube is Attached to the Boom External Container by a Marmon Clamp. The First Moveable Tube, Tube 1, is Driven by a Stepper Motor and Ball Screw Assembly.

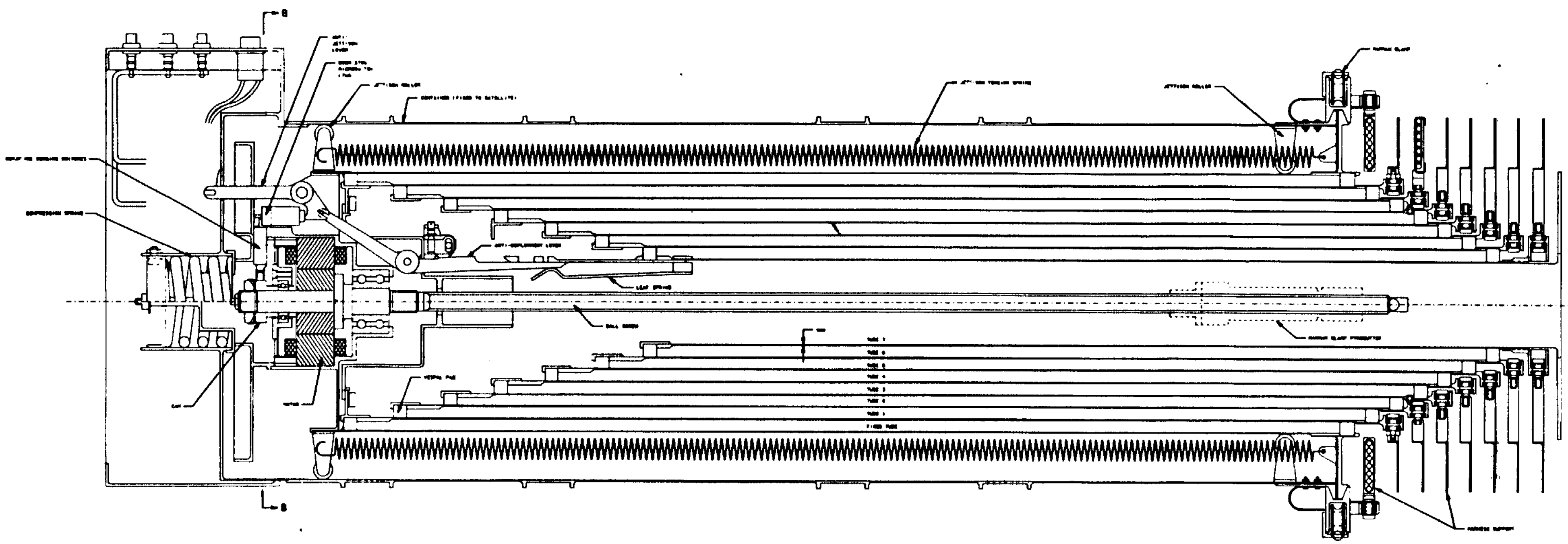
The Remaining 6 Tubes Are Deployed by Stainless Steel Cables Running on Roller Bearing Pulleys. Each Tube is Attached to its Consecutive Tube by 2 Redundant Sets of the Pulley/ Cable Assemblies. The Tubes are Extended Simultaneously by the Action of the Stepper Motor/Ball Screw Driving Tube 1. The Deploy Time is Approximately 5 Minutes.

The Tubes Slide on Vespel Pads and are Keyed Together with a Guide Sliding in Slots. The Electrical Harness for the Experiments is Mounted External to the Tube Assembly.

Each DRB Has 1 Two-Phase, Synchronous Stepper Motor.with a Constant Drive Torque of 0.36 N-m. The Motor is Current Limited, Draws 8W, and Rotates at 60 rpm

Deployable Retrievable Booms

Structure



Deployable Retrievable Booms

Latches

Each Boom Has 2 Independent, Redundant Mechanical Latches Which are Engaged When the Boom is Stowed. Each Latch Consists of 2 Pivoting Levers That Move Together and are Connected at a Common Pivot Point . 1 Lever is the Anti-Deploy Lever, the Other is the Anti-Jettison Lever.

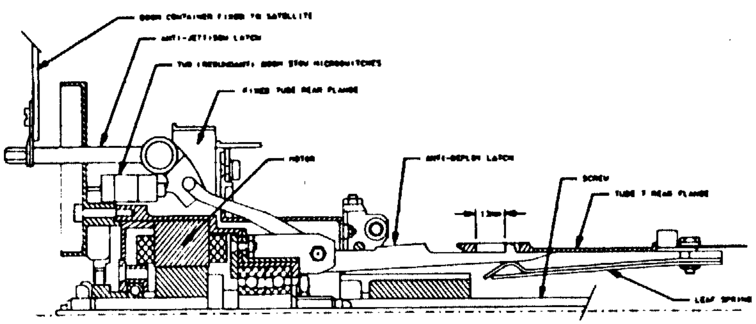
The Anti-Jettison Lever Hooks Onto the Fixed Structure of the External Container and Prevents Boom Jettison When the DRB is Latched. The Anti-Deploy Lever Fits in a Slot in a Flange Mounted to the 7th, or Inner, Tube. The Slot Allows for Thermal Expansion/Contraction of the Mechanism.

Both Latches Open after 2 mm of Motion of the DRB Drive Mechanism., A Roller Moving Axially along the Top of the Anti-Deploy Lever Moves over a Ramp, Forcing the Lever Down, and Disengaging It from the Flange in Tube 7. The Anti-Jettison Lever Disengages Simultaneously.

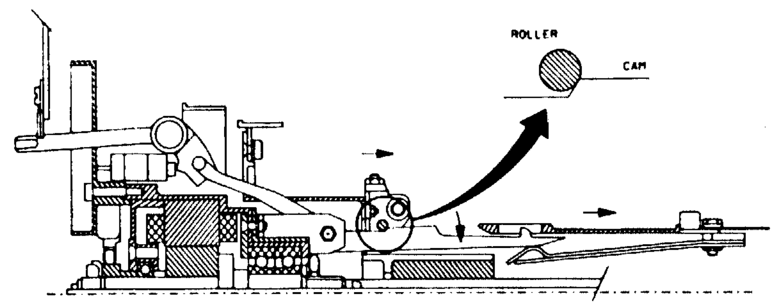
Deployable Retrievable Booms

Latches

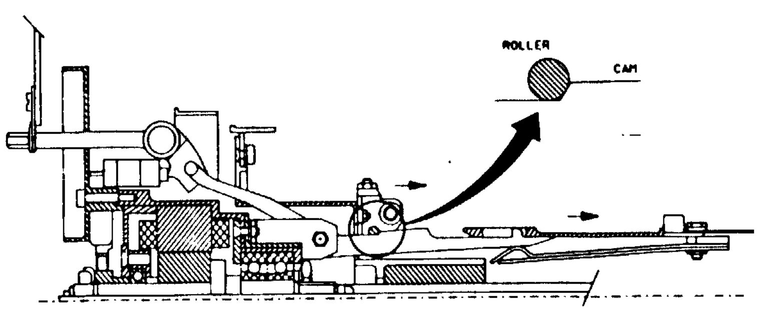
DEPLOY



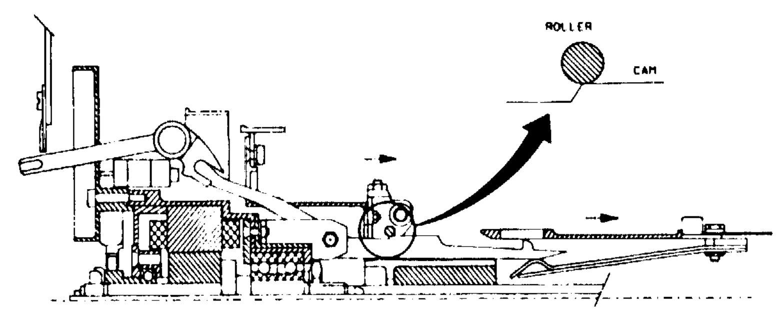
INITIAL POSITION BEFORE DEPLOYMENT



INTERMEDIATE POSITION DURING DEPLOYMENT



BEGINNING OF LATCH ROTATION DURING DEPLOYMENT



LATCHES FULLY OPEN DURING DEPLOYMENT

Deployable Retrievable Booms

Jettison System

Each Boom Has a Jettison Capability When Unlatched. The Jettison Sequence is Initiated by Firing 2 Redundant Pyro Cable Cutters, Mounted 180° Apart, to Sever the Marmon Clamps in 2 Places. Firing of 1 Cable Cutter is Sufficient to Ensure Proper Separation. The Marmon Clamp Cable Remains with the Jettisoned Portion of the DRB.

Initial Jettison Energy is Provided by 1 Spring Which Provides Sufficient Force to Demate the 2 Boom Electrical Connectors. 3 Tension Springs Spaced 180° Apart Impart a Jettison Velocity of 0.3 to 0.7 m/s. 1 Spring Will Provide 0.3 m/s Velocity, Which is Sufficient for Proper Separation. The Jettisoned Components Ride on 6 Rollers Located between the 1st Fixed Tube and the Jettisoned Boom.

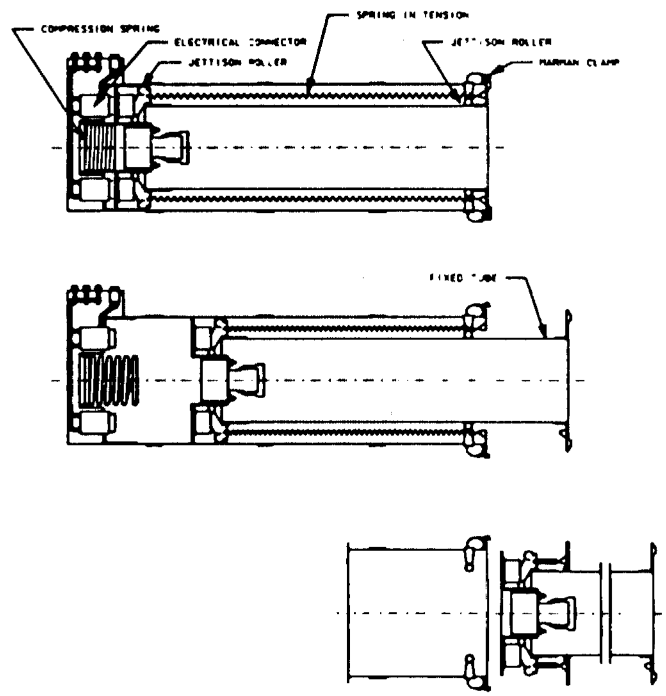
The DRB is Incapable of Jettisoning When Latched. The Anti-Jettison Lever React the Ejection Spring Loads if the Cable Cutters are Fired When the DRB is Latched. When the DRB is Unlatched, the Anti-Jettison Lever is Disengaged and Jettison is Possible.

The Jettison Pyrotechnics Electronics (JPE) Activates the Pyro Devices Which Provide Simultaneous Jettisoning of the 2 Booms. The DRB Jettison Function is Used Only if 1 or Both DRBs Fail to Retract and Latch Completely during Nominal Operations. The JPE is Powered Directly from the Satellite Main Bus. 2 Safety Inhibit Relays Within the JPE Prevent Accidental Jettison. Relay Status is Available to the Crew and MCC. These 2 Relays Must Be Commanded Closed Before the Fire Command is Issued. These Commands Are Available to the Crew and MCC.

When a JTS FIRE Command is Issued, a Sequencer Activates the 4 NSIs to Jettison Both Booms in 150 ms. The JPE Uses a 5-amp Solid State Current Limiter to Restrict the Current Drawn from the Satellite Main Bus during the NSI Firing. The Fire Pulse Duration is Approximately 25 ms.

Deployable Retrievable Booms

Jettison System



DRB Jettison Animation

Deployable Retrievable Booms

Electronics

Each DRB is Equipped with 5 Microswitches to Provide Status Information.

2 Microswitches Are Used to Verify That the DRB is Fully Retracted and Latched. 1 of the Microswitches is Wired to the DRB Electronics and is Used to Stop Motor Driving. The 2nd Switch is Wired Directly to the OBDH RTUP to Provide Status When the Motor is Powered Off.

2 Microswitches, Placed 90° Apart, Are Triggered by a Cam Mounted on the Ball Screw That Deploys the Boom Tubes. As the Screw Rotates, the Cam Triggers First One Microswitch, Then the Other. The Electronics Use the Microswitch Signals to Determine Screw Rotation Direction and Boom Position.

1 Microswitch is Mounted on the Tip of the External Container and is Triggered by the Jettisonable Portion of the Boom as it is Jettisoned. The Signal from This Microswitch Verifies That the Jettisoned DRB Has Cleared the External Container.

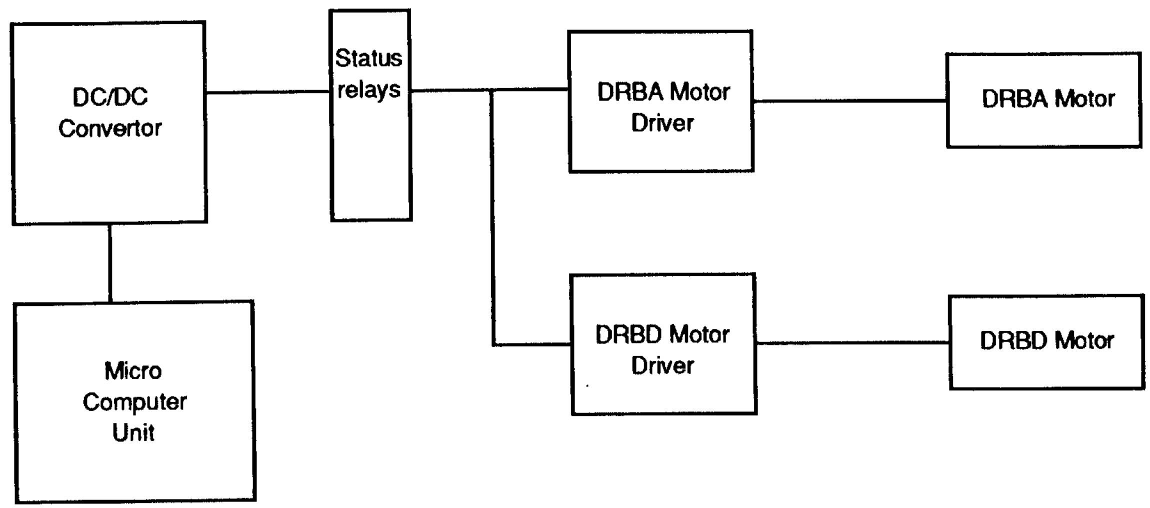
The DRB Electronics Unit (DRBE) Consists of the Motor Control Electronics (MCE) and the Jettison Pyrotechnics Electronics (JPE). The JPE Functions Were Described in the Boom Jettison Section.

The MCE Microcomputer is an 8086 CMOS, 16 Bit, 5 MHz Microprocessor with 8 kbytes of Program Memory PROM and Access to 4 kbytes of RAM.

The MCE Receives Power from the Satellite Main Bus and Delivers 28 Vdc to Each of the 2 Motor Driver Units. Within the Power Interface There are 3 Latching Relays Which are Used as Safety Inhibits to Prevent Accidental Boom Deploy. The 3 Inhibits Must Be Commanded Closed Before Issuing the Deploy Memory Load. To Deploy the DRBs, the Safety Inhibits are Commanded Closed, then a Memory Load Command is Sent. This Command Specifies the Deployed Length and Operational Mode.

Deployable Retrievable Booms

Electronics



6. Science

MPESS Science

Integration

The TSS-1 Science Mounted on the Mission Peculiar Experiment Support Structure Provided by the MSFC Spacelab Office is Illustrated in the Interface Block Diagram Below. Shown Are the Physical and Functional Interfaces.

The Cargo Bay Science Mounted on the MPESS Structure Includes:

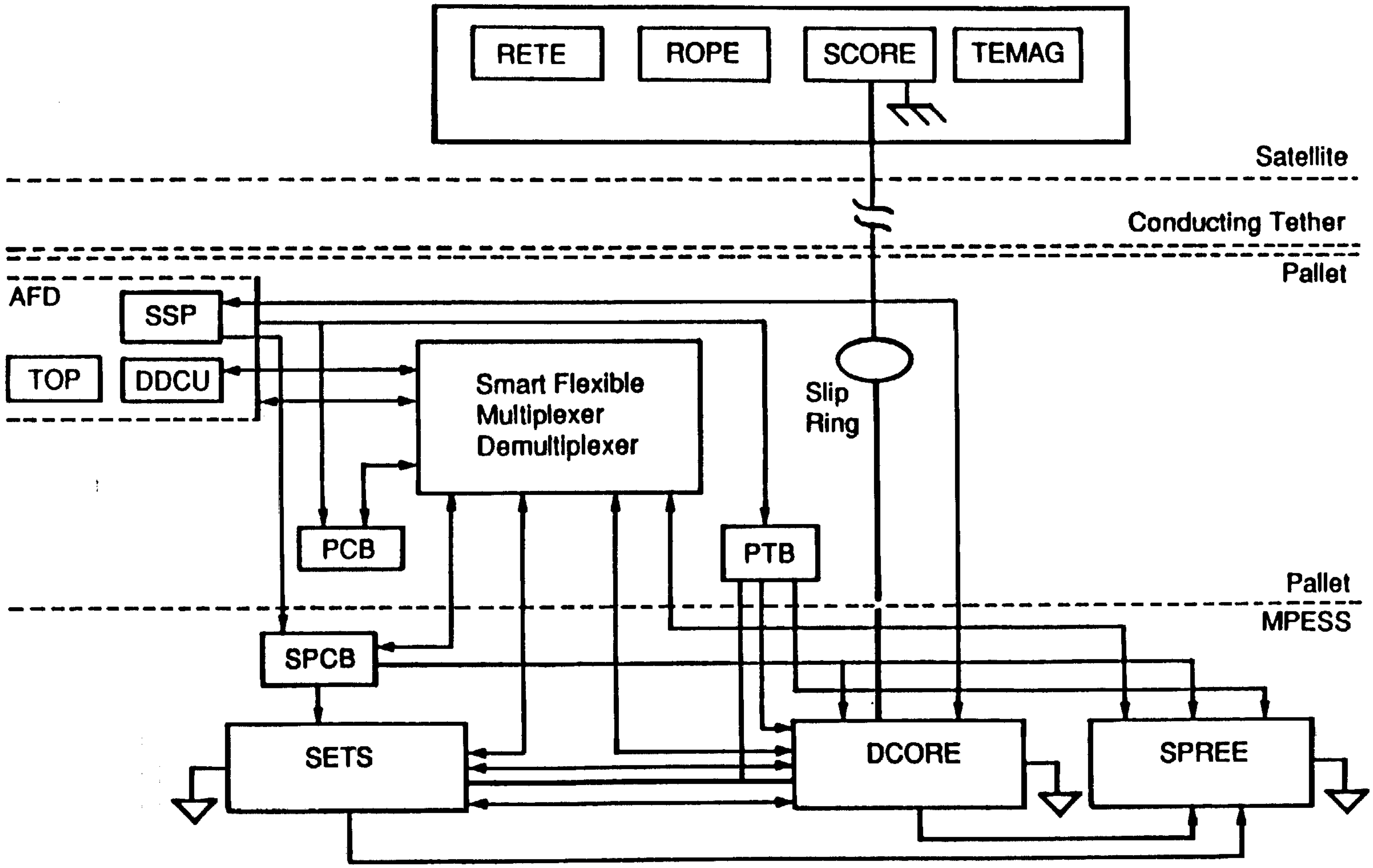
- Shuttle Electrodynamic Tether System (SETS)
- Shuttle Potential and Return Electron Experiment (SPREE)
- Deployer CORE (DCORE)

To Ensure the Compatibility of the System the Interface Requirements Are Documented in the Science Interface Agreements or Interface Control Documents.

The Resources Allocated through the Instrument Interface Agreements (IIAs) and Interface Control Documents (ICD) Include Structural Mechanical Requirements, Thermal or Environmental Control, Electrical Power, and Command and Data Management Interfaces Including Software and Functional Resource Allocation.

MPESS Science

Interface Block Diagram



MPESS Structure

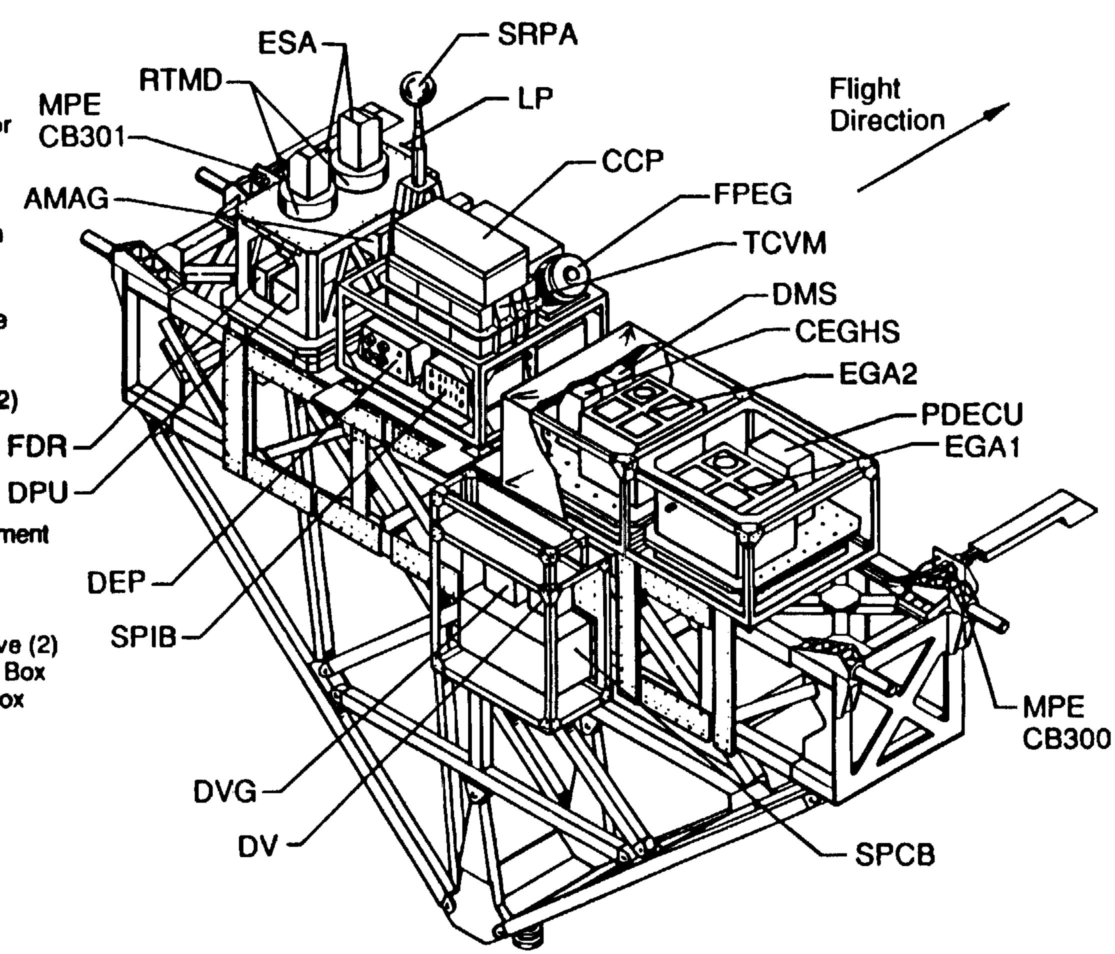
Description

The MPESS Carrier Provides Structural Support for the Cargo Bay Science Experiments and Mounting Accommodations for the Experiment Cold Plates, Electrical Cables, and Fluid Line Interfaces with the Deployer and Orbiter. The MPESS is a Riveted and Bolted Space Truss Assembly with Machined End Fittings Which Interface with the Orbiter Using Standard Spacelab Pallet Trunnions. The Structure is Fabricated from Aluminum Alloy and Assembled with Stainless Steel Fasteners.

MPESS Structure

Integrated MPESS

AMAG	Aspect Magnetometer
CB	Connector Bracket
CCP	Charge Current Probe
CEGHS	Core Electron Generator Head Switch
DEP	Dedicated Experiment Processor
DMS	Deployer Master Switch
DPU	Digital Processing Unit
DV	Deployer Voltmeter
DVG	Deployer Vacuum Gage
EGA	Electron Generator Assembly (2)
ESA	Electrostatic Analyzer (2)
FDR	Flight Data Recorder
FPEG	Fast Pulse Electron Generator
LP	Langmuir Probe
MPECB	Mission Peculiar Equipment Connector Bracket (2)
PDECU	Power Distribution and Electronic Control Unit
RTMD	Rotary Table Motor Drive (2)
SPCB	Science Power Control Box
SPIB	Sets Power Interface Box
SRPA	Spherical Retarding Potential Analyzer
TCVM	Tether Current Voltage Monitor



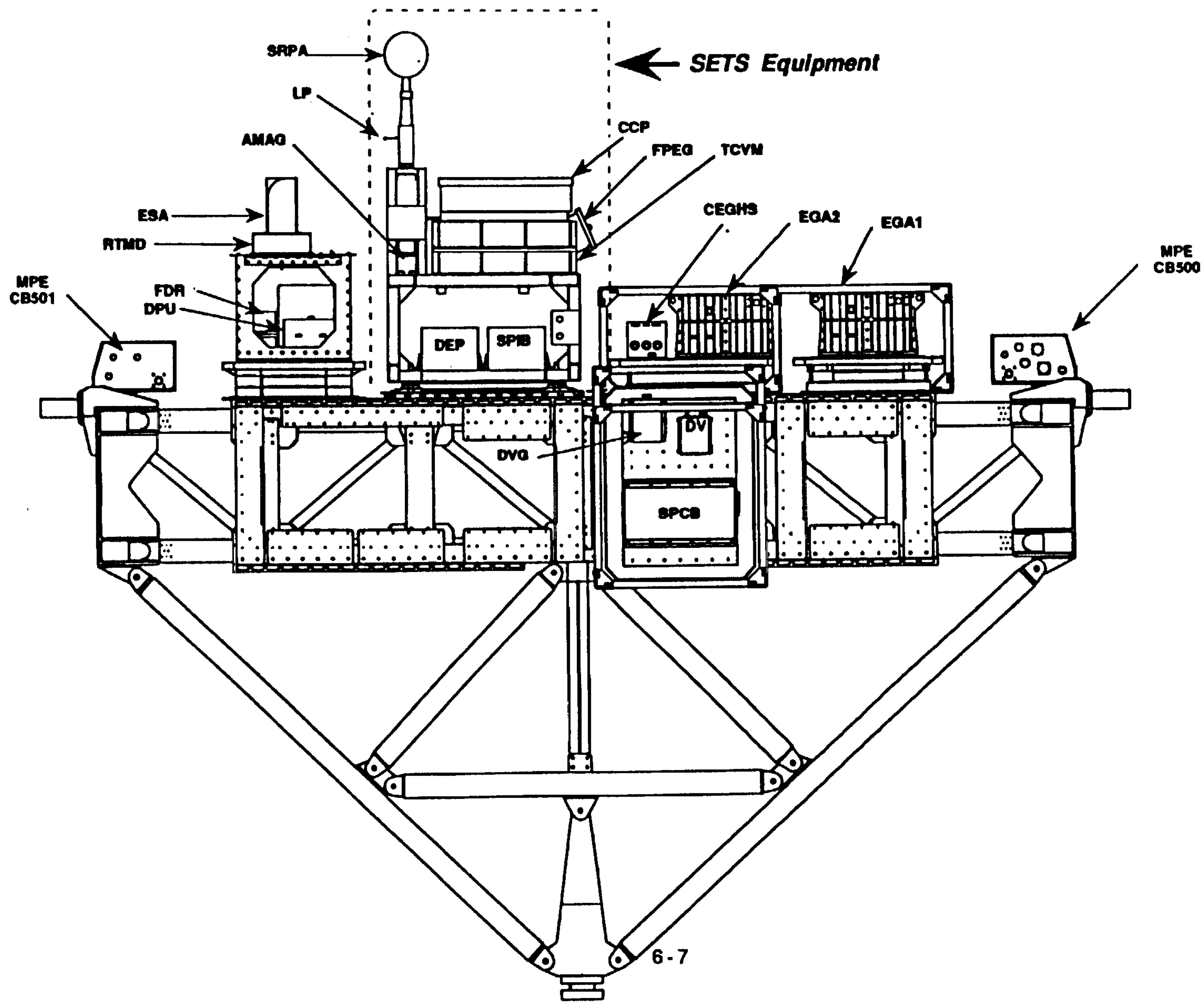
Shuttle Electrodynamic Tether System (SETS)

Mechanical Integration

The Shuttle Electrodynamic Tether System (SETS) Instrument is Integrated Onto the SETS Mounting Platform Before Delivery to the KSC Launch Site. The SETS Structural Interface to the MPESS is Via an MPE Adapter Plate. Two Right Handed Cold Plates Are Integrally Mounted to the SETS Mounting Platform (SMP) for Active Thermal Control. The two Cold Plates Are Connected in Series and Piped to an Interface Bracket Where the External Interface Will Be an Inlet and Exhaust Coolant Line.

SETS

Mechanical Configuration



SETS

Electrical Configuration

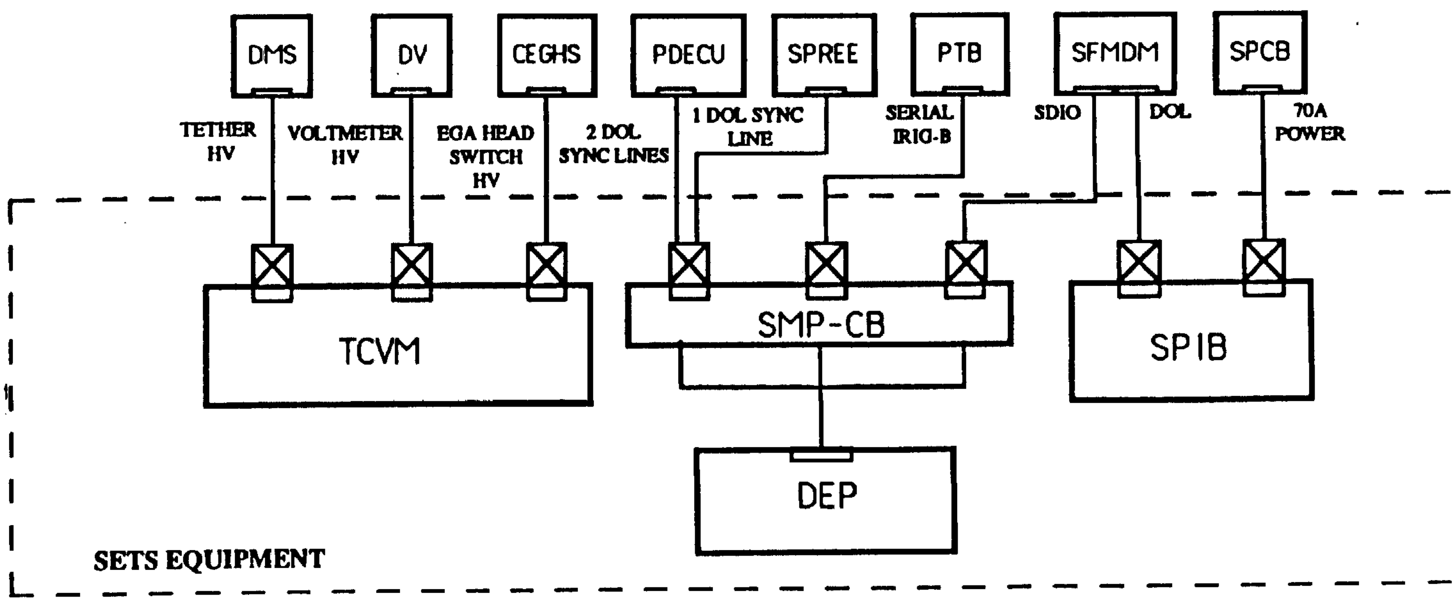
The Block Diagram Below Illustrates the SETS External and Internal Interfaces.

The Tether is Connected to the Tether Current Voltage Monitor (TCVM) via a High Voltage Coaxial Cable, Rated at 40 kV, from the Deployer Master Switch (DMS). The TCVM Utilizes a Hall Effect Device to Monitor the Tether Current.

Following the Current Monitor, 2 Outputs are Provided to the DCORE, 1 to the Deployer Voltmeter (DV) and 1 to the Core Electron Generator Head Switch (CEGHS). The DV, As Well As the CEGHS, Have the Capability to Disconnect Any Load on the Tether. 2 DOL Sync Commands are Provided by the DCORE Power Distribution and Control Unit (PDECU) to the SETS Dedicated Experiment Processor (DEP) to Indicate When the EGA is in the Beam-On Mode. The Tether Current Can Be Routed in the TCVM Directly to Orbiter Structure. The Electrons for the Fast Pulse Electron Generator (FPEG) Electron Beam are Taken Directly from the FPEG Chassis Which is Electrically Tied to the Orbiter Chassis. 1 Digital Synchronization Line to SPREE Indicates the Pulsing of the FPEG.

SETS

Electrical Configuration



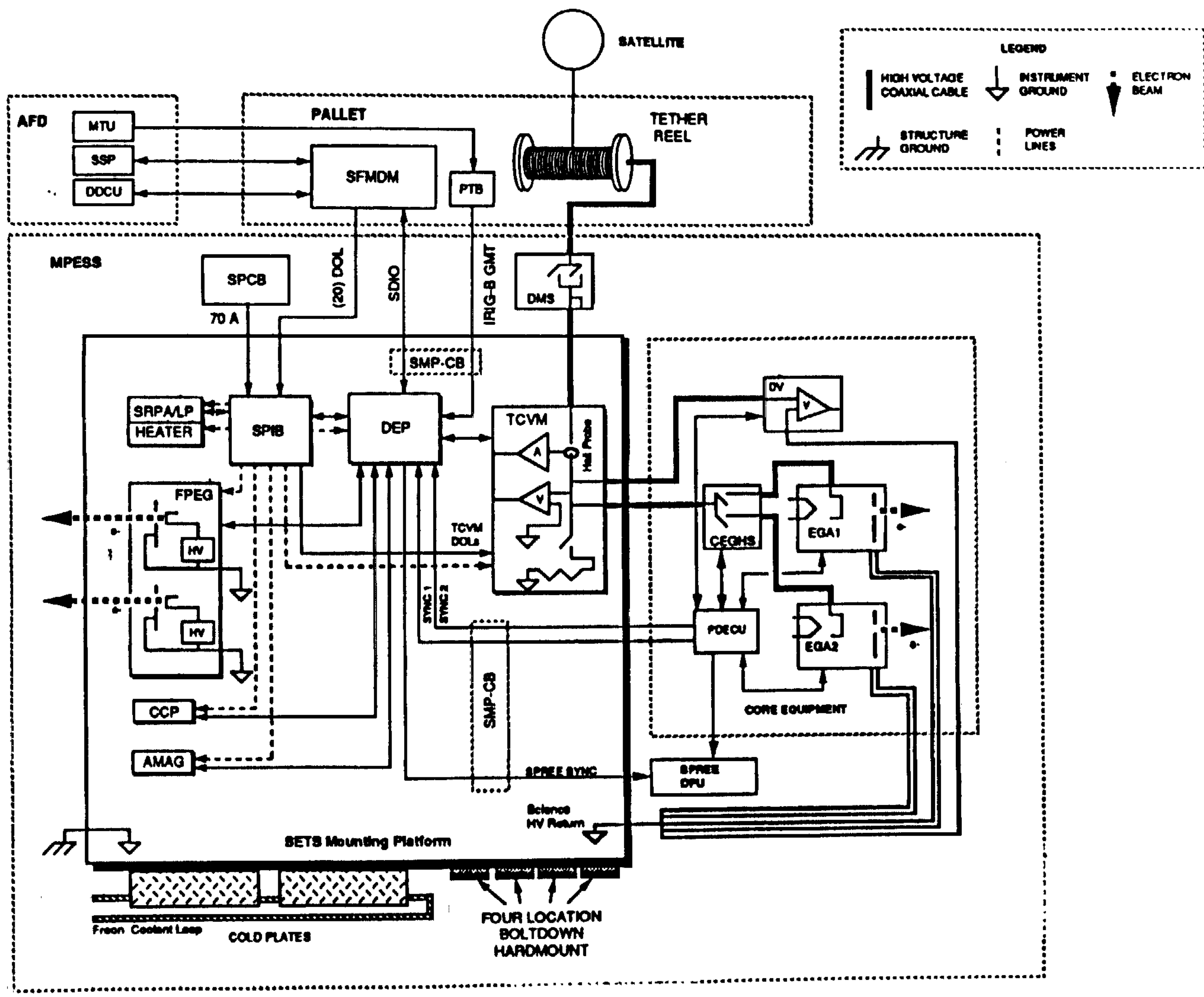
SETS

Electrical Configuration

A 70 Amp Power Input Is Provided to the SETS Power Interface Box (SPIB) from the Science Power Control Box (SPCB). As Illustrated in the Block Diagram Below, Several Interfaces are Identified between SETS and SPREE, the DCORE PDECU, DV, DMS and CEGHS, the EMP SFMDM.

SETS

Electrical Configuration



SETS

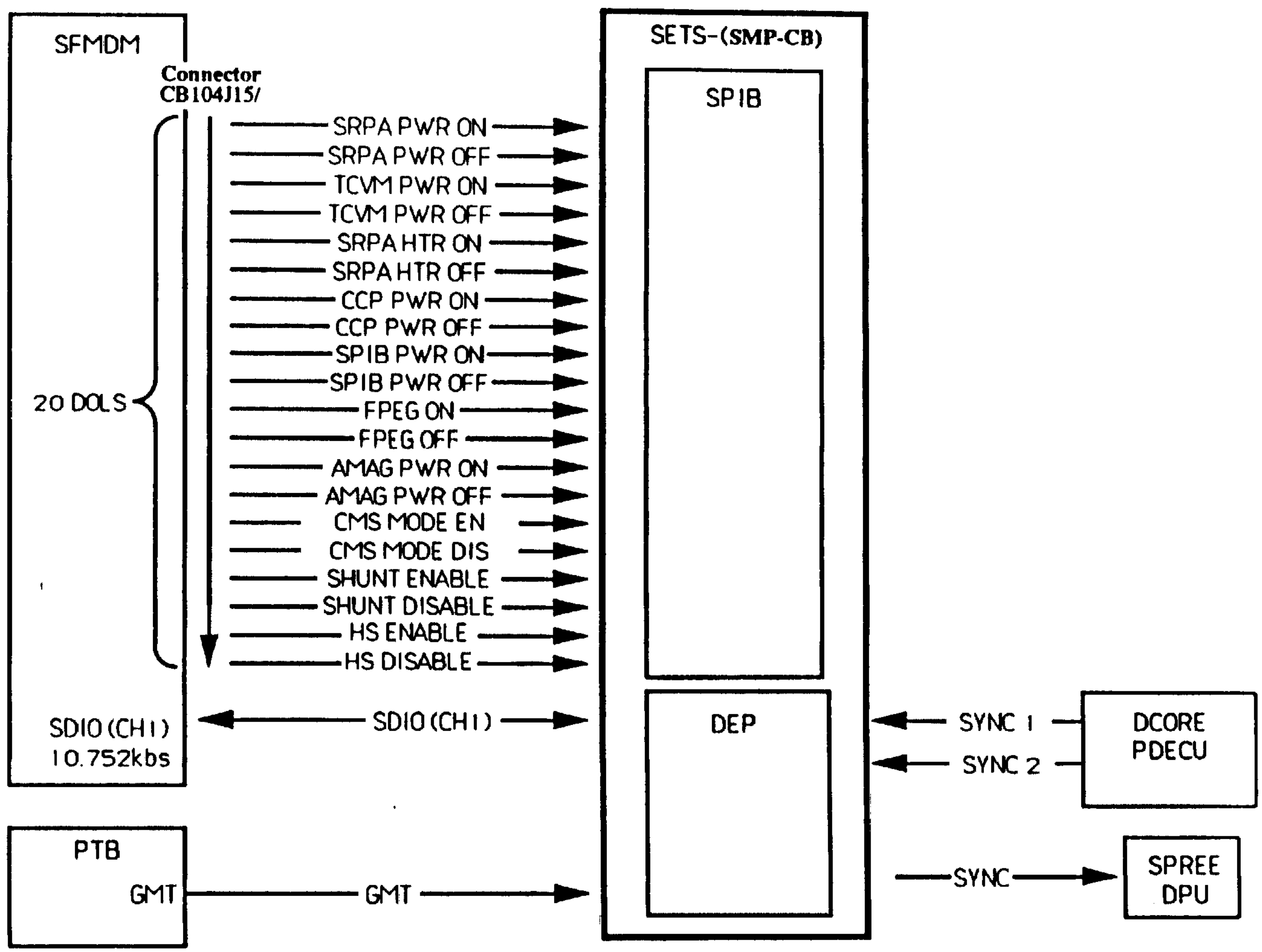
Command & Data Management Interfaces

Illustrated Below are the Command and Data Management Interfaces for the SETS Experiment.

The EMP SFMDM Supplies 20 DOL Command Inputs for Power Switching Relays in the SPIB for the Various Units Within the SETS Experiment. A Bi-Directional Serial I/O Port is Provided for Sending Commands from the SFMDM to the DEP and for Transmitting Data from the Dedicated Experiment Processor (DEP) to the SFMDM. Timing is Provided by an IRIG-B Encoded GMT Signal from the Payload Timing Buffer (PTB).

SETS

Command & Data Management Interfaces



Shuttle Potential and Return Electron Experiment (SPREE)

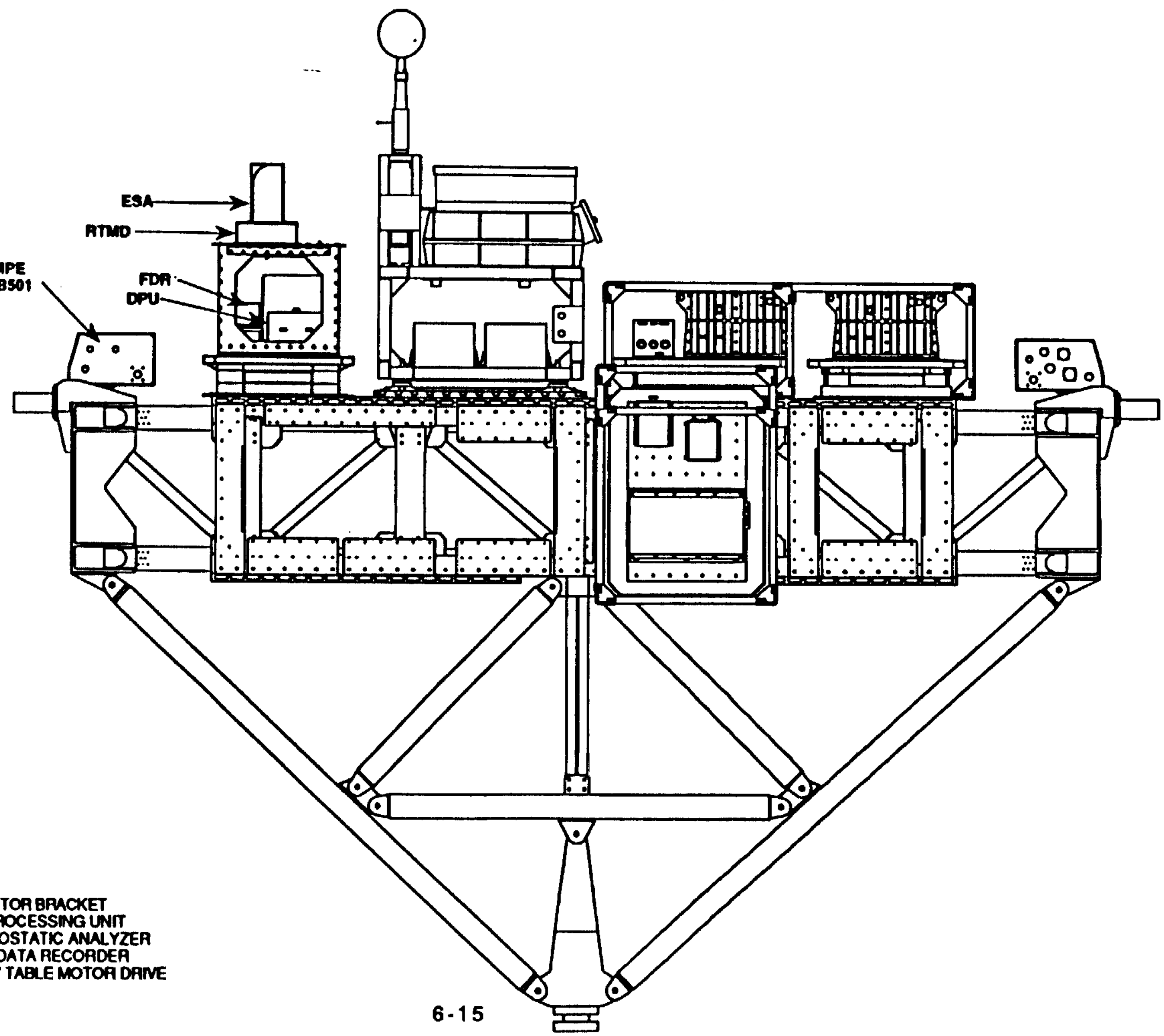
Mechanical Interfaces

The SPREE Instruments are Mounted on the SPREE Mounting Bracket Which Is Attached to the MPESS.

Environmental Control for SPREE is Provided through a Standard Spacelab Cold Plate and Associated Plumbing, Thermal Insulation and Thermal Interface Filters.

Shuttle Potential and Return Electron Experiment (SPREE)

SPREE Equipment



CB	CONNECTOR BRACKET
DPU	DATA PROCESSING UNIT
ESA	ELECTROSTATIC ANALYZER
FDR	FLIGHT DATA RECORDER
RTMD	ROTARY TABLE MOTOR DRIVE

SPREE

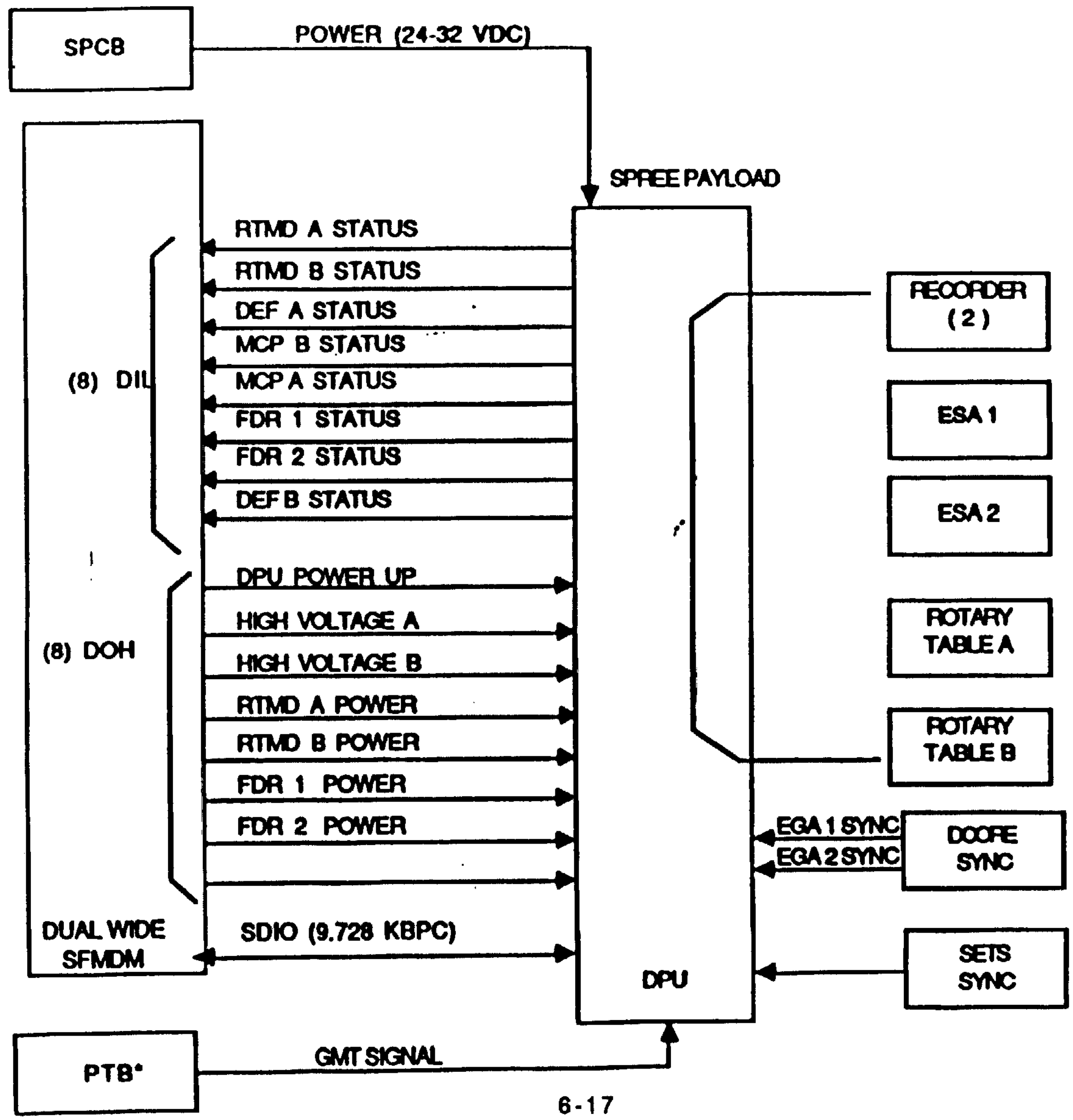
Electrical Interfaces

Orbiter Power Is Provided to SPREE at the Data Processing Unit (DPU) Power Connector from the MPESS SPCB. Power from the SPCB Is Fused with a 10 A Fuse. The Power Has a Dedicated Power Relay.

The Figure Below Illustrates the Electrical Interfaces between the SPREE Instrument , SETS, DCORE, and the EMP Elements.

SPREE

Electrical Interfaces



SPREE

Command & Data Management

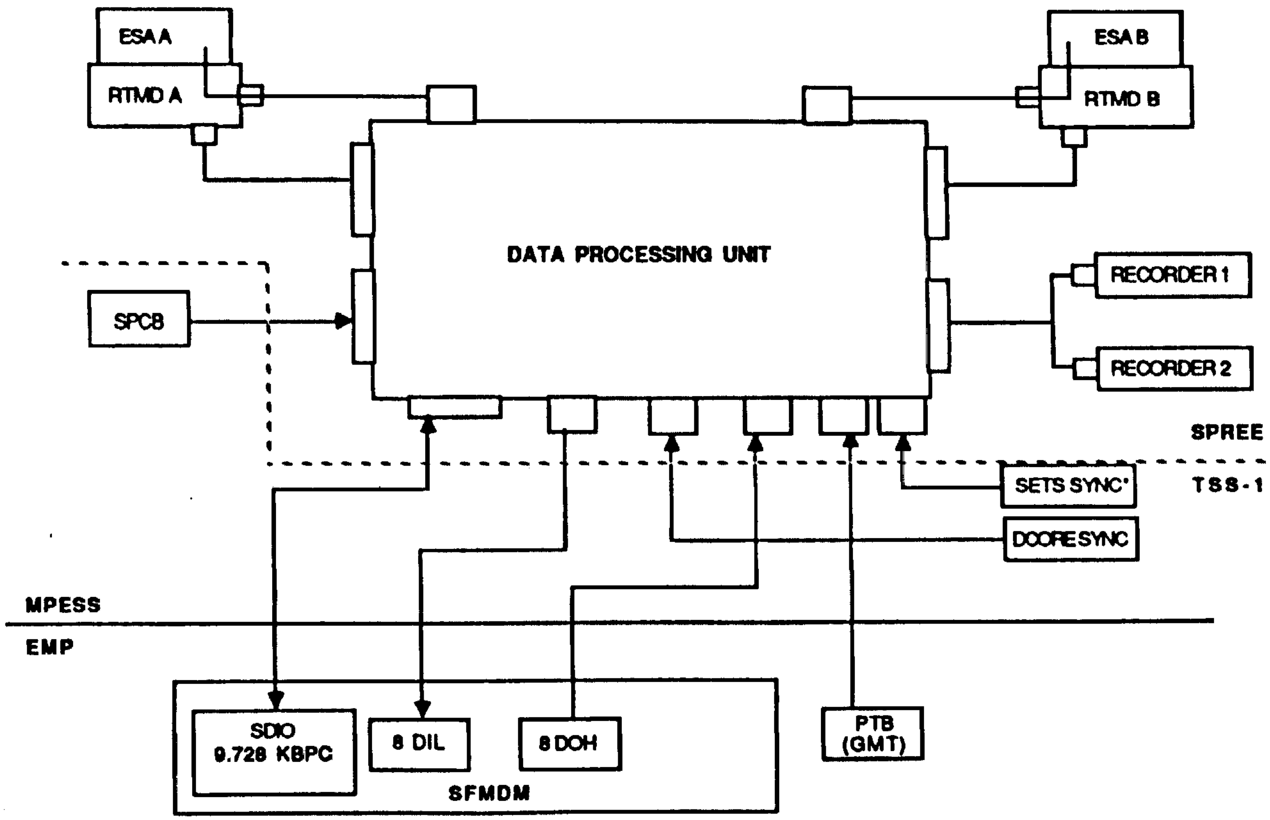
The SPREE Command and Data Management System (C&DMS) Interfaces with the SFMDM, the SETS, and the DCORE.

The SFMDM Provides Commands to the DPU via a Single Serial I/O Channel. This Channel Also Serves as the Telemetry Data Channel for the Data Output by the DPU.

The DPU Also Utilizes 4 High Level Discretes for Commanding, and 8 Low Level Discretes for Monitoring the SPREE Instruments. The DPU Receives 3 Low Level Discrete Synchronization Signals for Auto-Correlation of the SPREE Wave Particle Correlation, 1 from the SETS and 2 from the DCORE.

SPREE

Command & Data Management



Deployer Core Equipment (DCORE)

Equipment Description

The Deployer Core Equipment (DCORE) Consists of 7 Components:

The Deployer Master Switch (DMS) Electrically Connects and Disconnects the Conductive Tether from the Payload Bay Science Equipment.

The Power Distribution and Electronic Control Unit (PDECU) Provides Power Commanding, and Data Interfacing Electronics for the DCORE Equipment, Except for the DMS. The PDECU Also Provides Electron Generator Synchronization Signals to SETS and SPREE Experiment Equipment.

The Deployer Vacuum Gauge (DVG) Measures the Ambient Gas Pressure to Prevent EGA Operation Under Pressure Conditions Which May Cause Arcing.

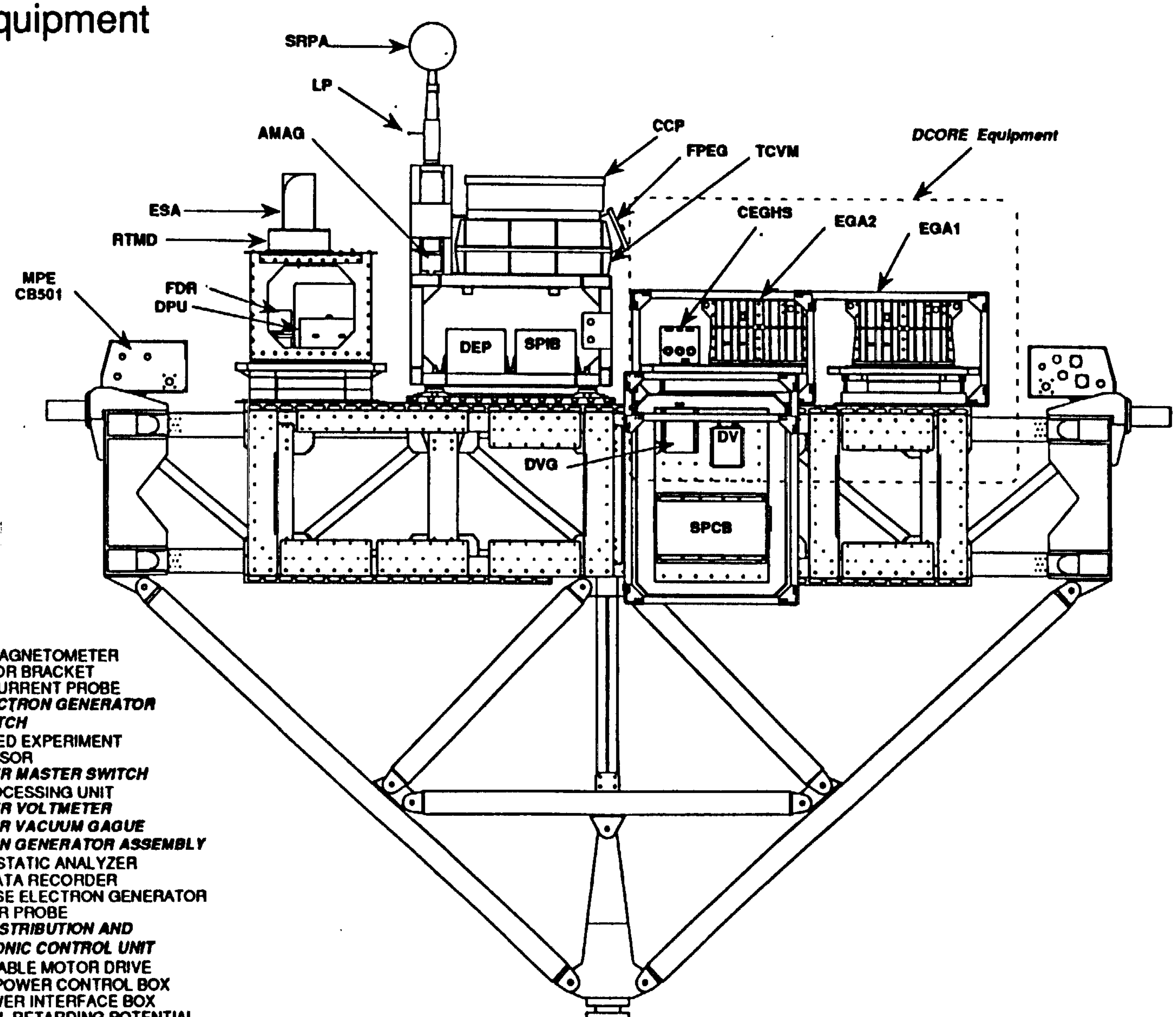
The Deployer Voltmeter (DV) Measures the Tether Electrical Potential Relative to the Orbiter Structure.

The 2 Electron Generator Assemblies (EGA) Control the Current Flowing Down the Tether by Emitting the Current Via an Electron Beam Utilizing the Electromotive Force Induced across the Tethered Satellite.

The Core Electron Generator Head Switching Device (CEGHS) Allows Connection and Disconnection of the Tether Voltage to Either of the 2 CORE Electron Generator Cathodes.

DCORE

DCORE Equipment



AMAG	ASPECT MAGNETOMETER
CB	CONNECTOR BRACKET
CCP	CHARGE CURRENT PROBE
CEGHS	***CORE ELECTRON GENERATOR HEAD SWITCH***
DEP	DEDICATED EXPERIMENT PROCESSOR
DMS	***DEPLOYER MASTER SWITCH***
DPU	DATA PROCESSING UNIT
DV	***DEPLOYER VOLTMETER***
DVG	***DEPLOYER VACUUM GAGUE***
EGA	***ELECTRON GENERATOR ASSEMBLY***
ESA	ELECTROSTATIC ANALYZER
FDR	FLIGHT DATA RECORDER
FPEG	FAST PULSE ELECTRON GENERATOR
LP	LANGMUIR PROBE
PDECU	***POWER DISTRIBUTION AND ELECTRONIC CONTROL UNIT***
RTMD	ROTARY TABLE MOTOR DRIVE
SPCB	SCIENCE POWER CONTROL BOX
SPIB	SETS POWER INTERFACE BOX
SRPA	SPHERICAL RETARDING POTENTIAL ANALYZER
TCVM	TETHER CURRENT VOLTAGE MONITOR

DCORE

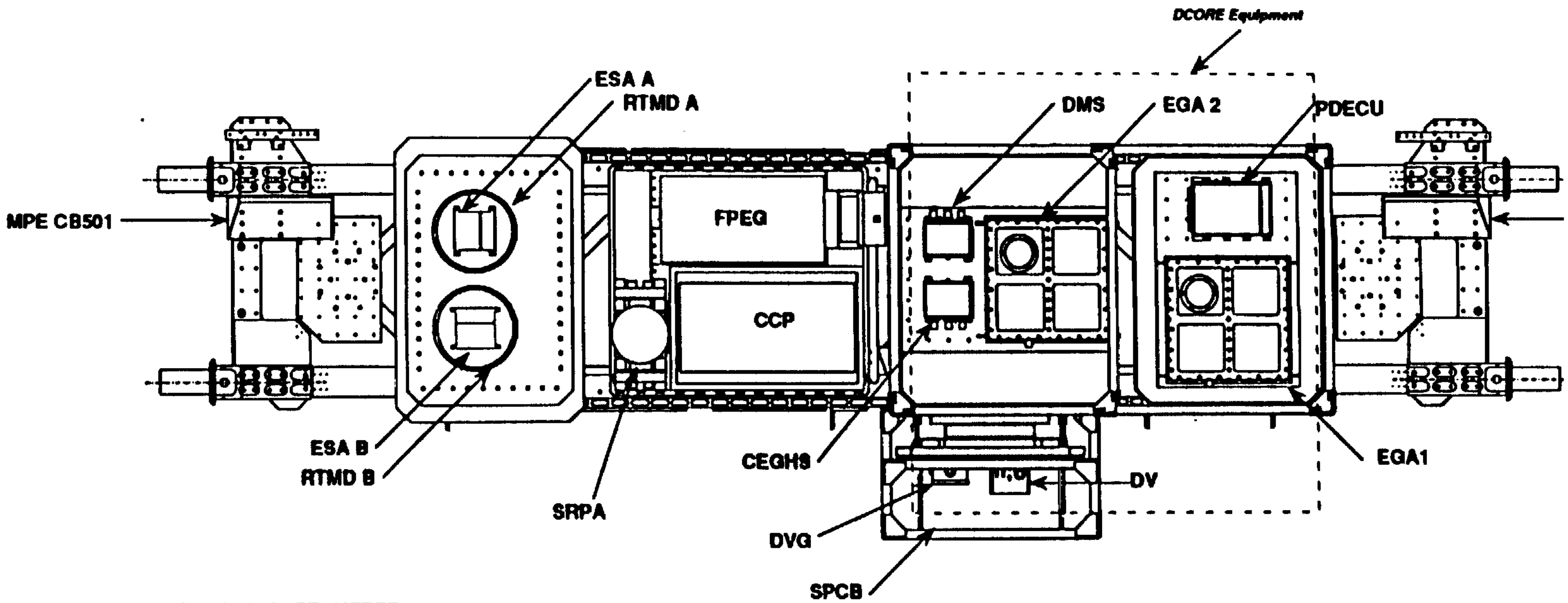
DCORE Thermal Control

The DCORE Units/Subassemblies Interface with 3 MPESS Mounted Spacelab Cold Plates. Each of the 2 EGA Units Are Mounted on Separate Cold Plates.

All DCORE Units/Subassemblies are Thermally Coupled to the Cold Plates with a Thermally Conductive Layer. The Cold Plate Mounted Units are Covered by MMAG Provided Thermal Blankets, with the Exception of EGA Heads that have Beam Escape Aperture, and a DVG Tube Which Extends Outside of the MLI Tent.

DCORE

Hardware Configuration



AMAG	ASPECT MAGNETOMETER
CB	CONNECTOR BRACKET
CCP	CHARGE CURRENT PROBE
CEGHS	*CORE ELECTRON GENERATOR HEAD SWITCH*
DEP	DEDICATED EXPERIMENT PROCESSOR
DMS	*DEPLOYER MASTER SWITCH*
DPU	DATA PROCESSING UNIT
DV	*DEPLOYER VOLTMETER*
DVG	*DEPLOYER VACUUM GAGUE*
EGA	*ELECTRON GENERATOR ASSEMBLY*
ESA	ELECTROSTATIC ANALYZER
FDR	FLIGHT DATA RECORDER
FPEG	FAST PULSE ELECTRON GENERATOR
LP	LANGMUIR PROBE
PDECU	*POWER DISTRIBUTION AND ELECTRONIC CONTROL UNIT*
RTMD	ROTARY TABLE MOTOR DRIVE
SPCB	SCIENCE POWER CONTROL BOX
SPIB	SETS POWER INTERFACE BOX
SRPA	SPHERICAL RETARDING POTENTIAL ANALYZER
TCVM	TETHER CURRENT VOLTAGE MONITOR

DCORE

Electrical Power

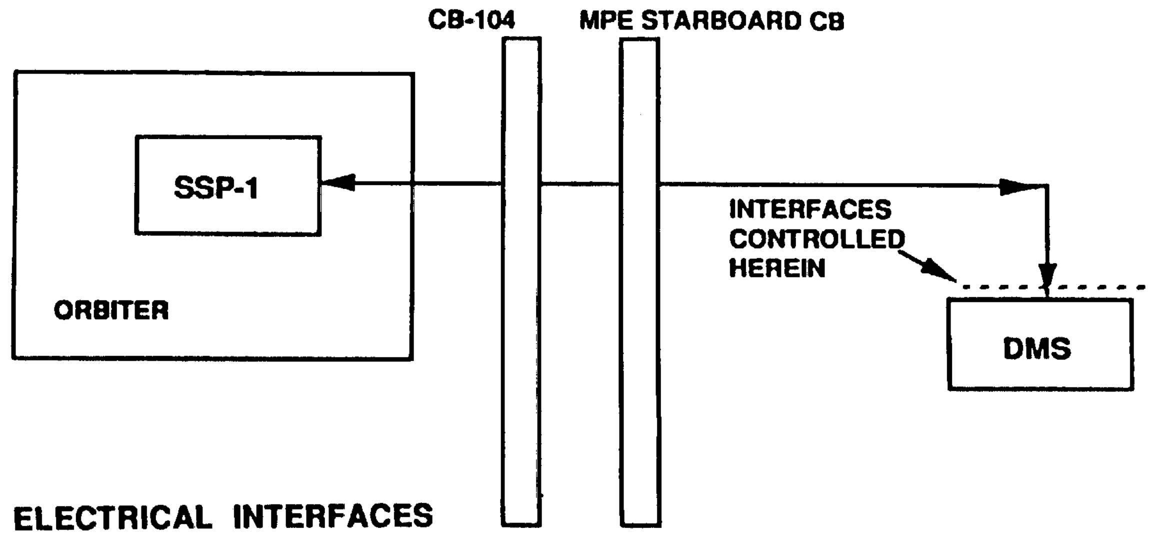
The DCORE Receives Orbiter Electrical Power from the SPCB with the Exception of the DMS. The DMS Is Powered through the Orbiter Standard Switch Panel (SSP). The DCORE PDECU Receives 1 28 Vdc Power Line from SPCB Capable of Carrying a Continuous Maximum Current of 70 Amps (100 A Fuse). The Power is Controlled on the Orbiter by a Switch Operated by External Commands and Monitored by External Monitors.

The DCORE DMS Receives 28 Vdc Power which is Crew Commanded and Monitored from the SSP.

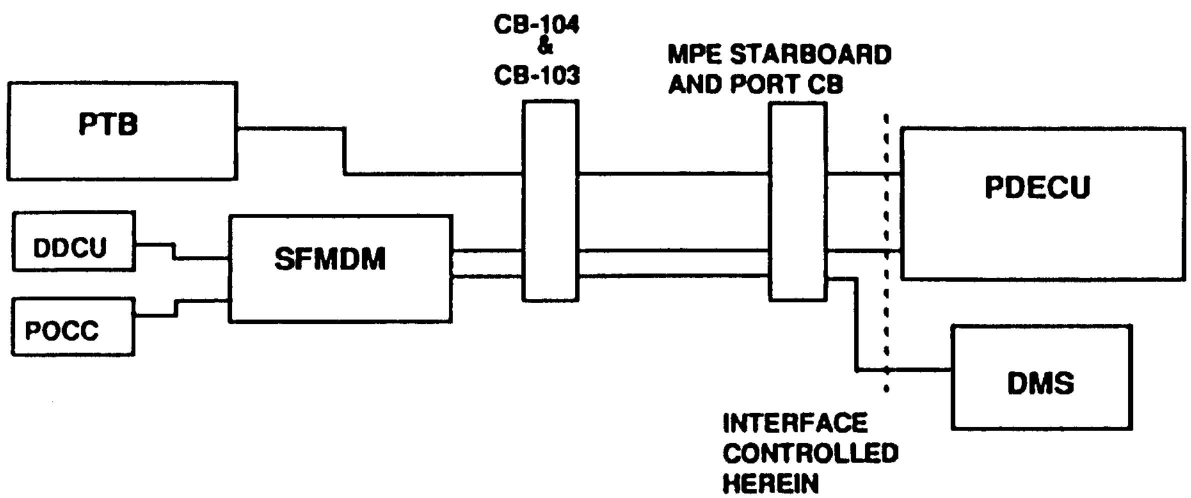
DCORE

Electrical Power

ORBITER TO DCORE ELECTRICAL INTERFACES



EMP TO DCORE ELECTRICAL INTERFACES



DCORE

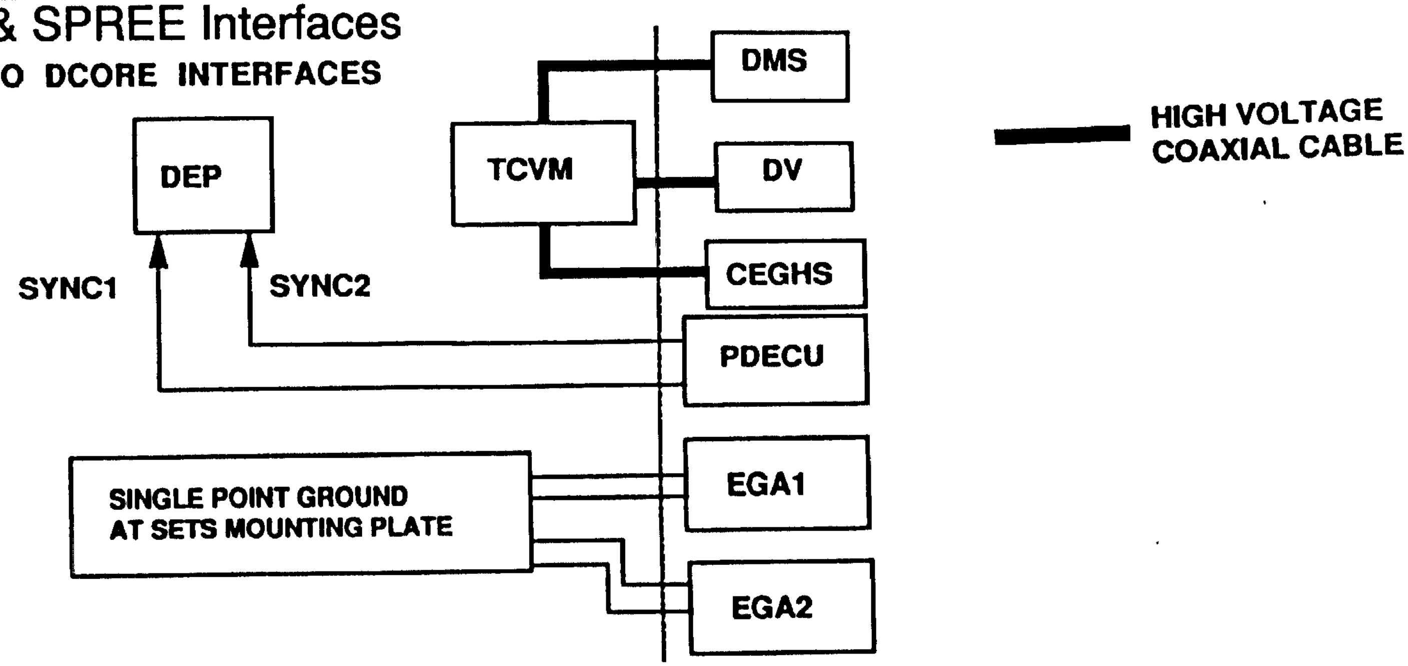
SETS & SPREE Interfaces

The DCORE Provides 4 Low Level Discrete Synchronization Signals to Indicate When the EGAs Are in an Electron Beam Emitting Mode. 2 Synchronization Signals Are Provided to SETS and 2 to SPREE.

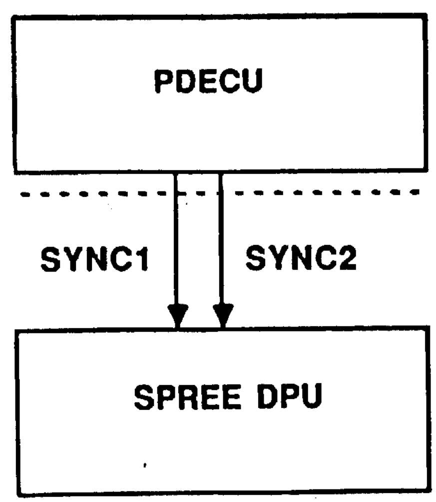
DCORE

SETS & SPREE Interfaces

SETS TO DCORE INTERFACES



SPREE TO DCORE INTERFACES



Satellite Mounted Science

Overview

Satellite Mounted Science Consists of the Satellite Core Experiment (SCORE), the Research on Electrodynamics Tether Effects (RETE), the Research on Plasma Electrodynamics (ROPE), and the Tether Magnetic Field (TEMAG).

The Satellite Core Experiment (SCORE) is Mounted Inside the Satellite Body.

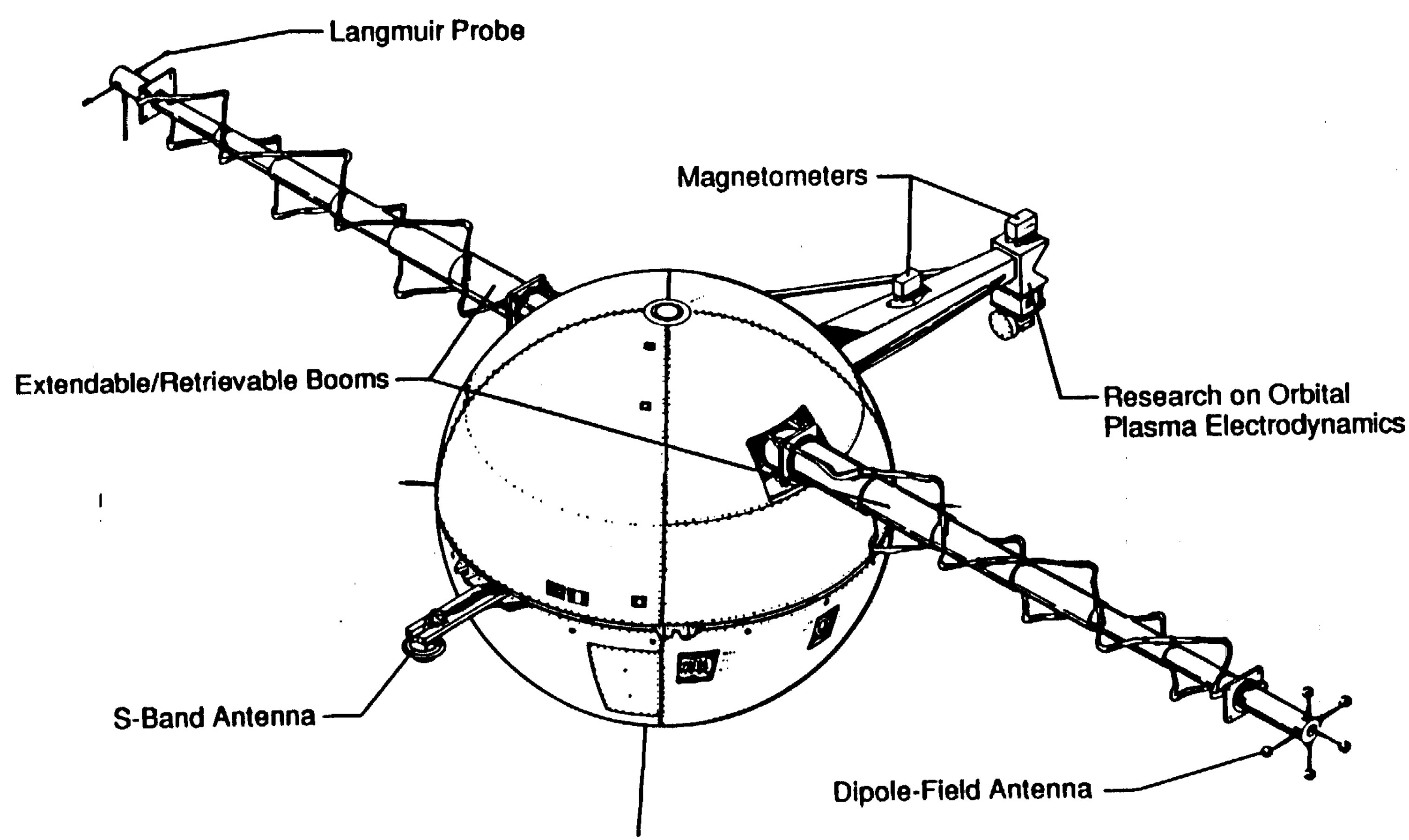
The Research on Electrodynamics Tether Effects (RETE)is Mounted on the DRBs.

The Research on Plasma Electrodynamics (ROPE) is Mounted on and Within the Satellite Body.

The Tether Magnetic Field (TEMAG) Has Experiment Sensors on the Fixed Boom and the Satellite Body.

Satellite Mounted Science

Overview



Satellite Core Experiment (SCORE)

Description

The SCORE Experiment Has 2 Components, the Satellite Ammeter (SA) and the Satellite Linear Accelerator. The Objectives of the SCORE Instrumentation Are to Provide Data on Dynamic Disturbances, to Help Assess TSS Current Collection, and to Support Other Data Interpretation.

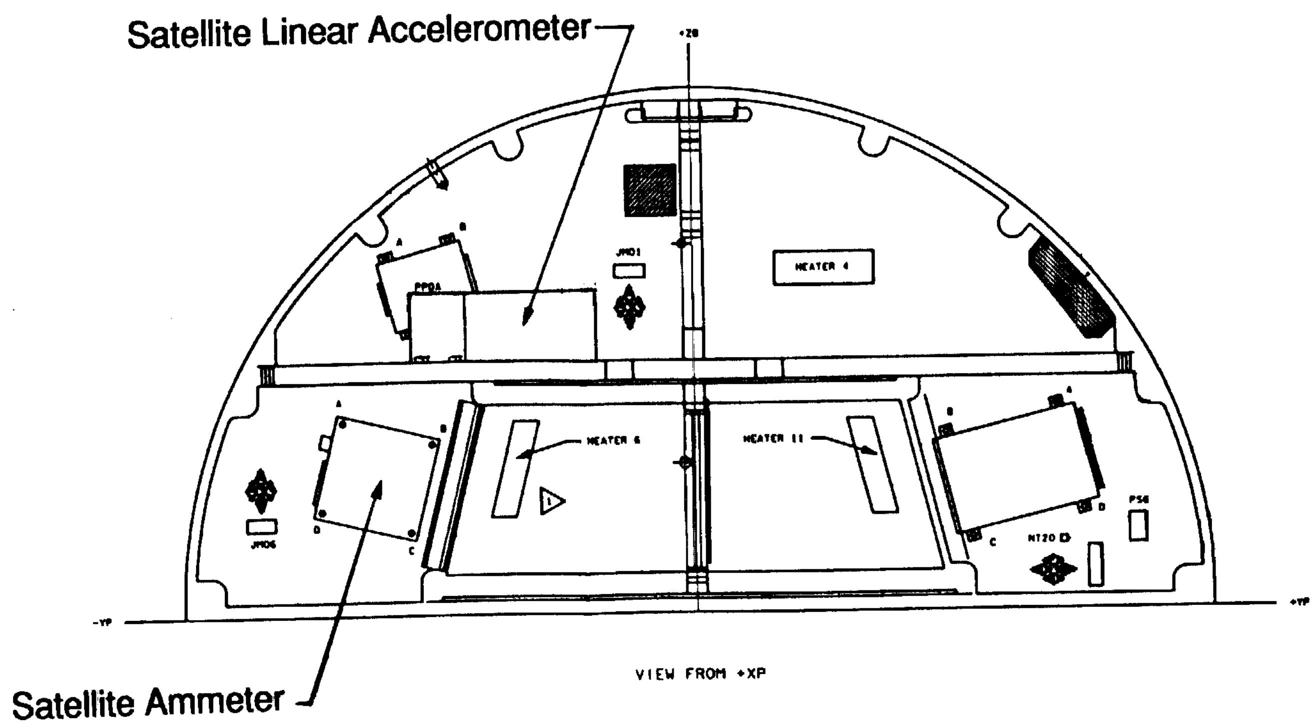
The Satellite Linear Accelerometer (SLA)is a 3-Axis Accelerometer That Measures Acceleration along 3 Orthogonal Axes. The Accelerometers are a Low-g, Pendulous Design. The SLA Uses the Balance between the Acceleration Forces Acting on a Proof Mass of a Mechanical Spring and the Electromagnetic Force of a Rebalancing Coil. Proof Mass Motion from an External Acceleration is Detected by a Capacitance-Measuring Sensor That Provides an Error Signal. This Signal Generates the Proper Input Current to the Balancing Coil and Returns the Proof Mass to Its Null Position. The Balancing Coil Current is Then Converted to a Voltage by a Resistor. This Voltage is Proportional to the Acceleration Input. The SLAs Output is Fed Directly to the Satellite OBDH.

The Accelerometers Require a Warm-Up to 69°C to Achieve Thermal and Mechanical Stability. This Warm-Up Takes Approximately 20 Hours.

The Satellite Ammeter (SA) is a Resistor-Type Current Meter That Measures Tether Current. It is an Autoscaling Instrument with Different Performances Depending on the Type of Output.

The SA Provides Housekeeping Data, Including Scale and Health Status, and a 16-bit Serial Current Measurement. Tether Current Values Are Also Sent from the SA to the TEMAG and RETE Experiments for Use.

SCORE Equipment



VIEW FROM +XP

Research on Electrodynamics Tether Effects (RETE)

Description

The RETE Experiment consists of 2 Packages, the AC Package and the Plasma DC Package, 1 Mounted on Each DRB.

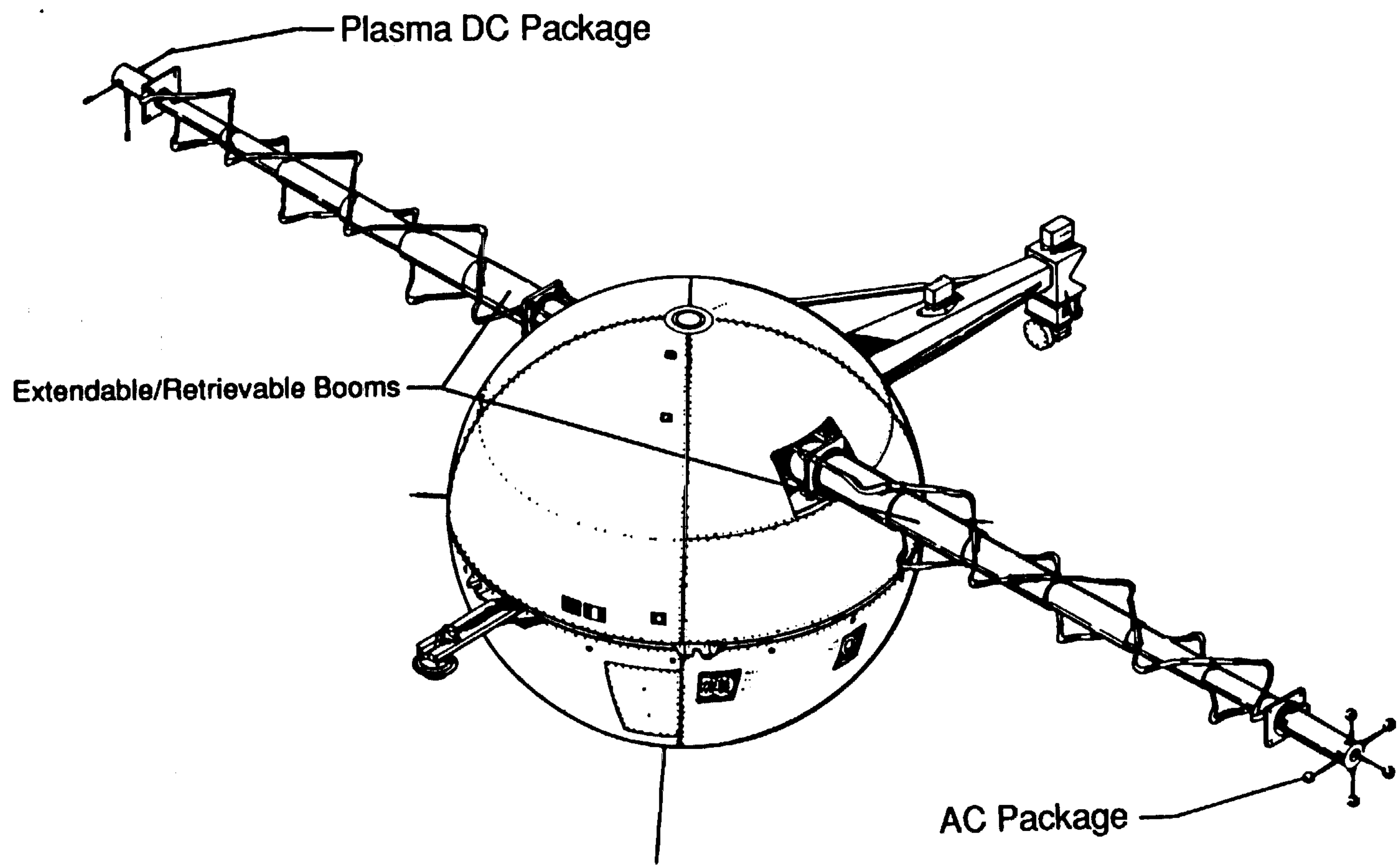
The AC Package Measures Fluctuations in the Electrical and Magnetic Fields Surrounding the Satellite. Electrical Fluctuations May Be Measured in All 3 Axes and at Frequencies Ranging from 200 Hz to 12 MHz. Magnetic Field Fluctuations Can Be Measured in the x and z Directions Only with Frequency Measurements from DC to 500 KHz. Magnetic Field Frequency Measurements Up to 12MHz Can Be Made, But Only in the z Direction.

The Plasma DC Package Analyzes the Sheath of Charged Particles That Surround the Satellite As It Passes through the Earth's Magnetosphere. The DC Package Houses 3 Langmuir Probes and an Ion Sensor for This Purpose. These Devices Are Used to Study the Electric Field Around the Satellite As Well As the Energy and Direction of Ions and Electrons in the Charged Sheath. The Electrical Field from DC to 200 Hz Is Also Analyzed by This Package.

Commands and Telemetry for RETE Are Routed by the Satellite OBDH. From the OBDH, Commands for the AC Boom Package (ACBP) Are sent to the Data Processing Unit (DPU). Commands for the DC Boom Package (DCBP) Are Sent to the DC Package Electronics (DCE) Unit.

Research on Electrodynamics Tether Effects (RETE)

RETE Equipment



Research on Plasma Electrodynamics (ROPE)

Description

The ROPE Experiment is Designed to Help Gain an Understanding of TSS Electrodynamic Interactions with the Magnetoplasma of the Earth's Ionosphere.

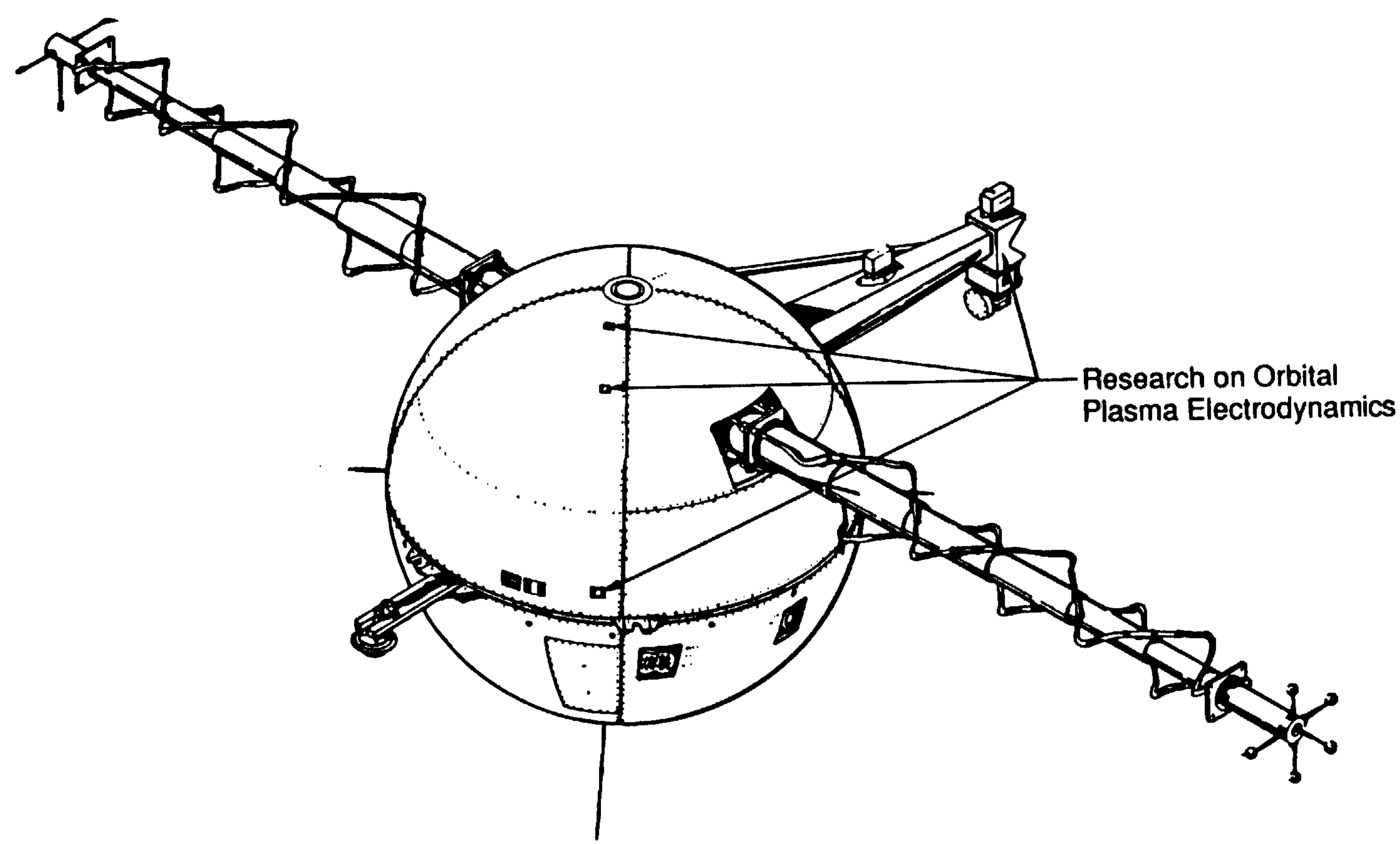
ROPE Uses Sensors Mounted on the Satellite Skin and on the Fixed Satellite Boom (EDY Boom). These Sensors Determine Ion Flux Density, Temperature, Direction of Travel, and Energy. The Package on the EDY Boom is the Boom-Mounted Sensor Package (BMSP). The ROPE Sensors Include the Differential Ion Flux Probe Sensors (DIFP-S) and the Soft Particle Energy Spectrometer (SPES).

The DIFP-S Uses 2 Electrical Sweeps to Determine the Angle of Attack, Temperature, and Flux Density of Incoming Ions. The DIFP-S is Located on the Fixed Electrodynamics (EDY) Boom As Part of the BMSP.

The SPES Consists of 5 Sensor Heads. 2 of These are Mounted on the EDY Boom as Part of the BMSP. The Other 3 are Mounted at 0°, 45°, and 90° to the Satellite Spin Axis on the Surface of the Satellite. These Sensors are Used to Measure the Energy Distribution of Ions and Electrons in the Vicinity of the Sensors.

Research on Plasma Electrodynamics (ROPE)

ROPE Equipment



Tether Magnetic Field Experiment (TEMAG)

Description

The TEMAG Sensors Provide Sensitive Measurements of Magnetic Field Variations as the TSS Orbits the Earth. The TEMAG Experiment Uses 2 Fluxgate Magnetometers (FGMs) Mounted on the EDY Boom. The Outermost FGM is Referred to As the Outer-Gradient Magnetometer, and the Innermost Is the Inner-Gradient Magnetometer. The Majority of the TEMAG Science Data Will Result from the Outer-Gradient FGM. The Inner-Gradient FGM, Because it is Closer to the Satellite Skin, Will Provide Real-Time Estimates on the Effects of the Satellite Body. Inner-Gradient FGM Readings Will Be Used Mostly for Post Flight Data Reduction.

Tether Magnetic Field Experiment (TEMAG)

TEMAG Equipment